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RESEARCH MEMORANDUM

ANALYTICAL COMPARISON OF TURBINE-BLADE COOLING SYSTEMS DESIGNED FOR A TURBOJET ENGINE OPERATING AT SUPERSONIC SPEED AND HIGH ALTITUDE

I - LIQUID-COOLING SYSTEMS

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SUMMARY

The influence of high-altitude supersonic flight on the operation and effectiveness of turbine-blade liquid-cooling systems for application in turbojet engines in guided missiles and in supersonic aircraft was investigated analytically. The turbine blades in such applications must be effectively cooled to obtain the increased engine air-handling capacity, blade tip speed, and turbine-inlet gas temperature required for adequate specific engine thrust and weight. The problems encountered in liquid-cooling systems were investigated with reference to several specific designs for alternate heat-rejection mediums, and the results are presented herein.

Results of the analysis for liquid-cooling systems showed that sufficiently low blade temperatures could be obtained to provide adequate blade strength for the type and size of engine currently of interest for interceptor application. The water-cooling system appears inadequate for an interceptor mission at Mach numbers up to 2.5 at 50,000 feet altitude for a turbine-inlet temperature of 2040° F because of its inability to reject heat to ram air at high flight Mach number and because of considerable installation complexity. A fuel-cooling system with heat rejection to the afterburner fuel appears promising because of negligible weight and performance penalties and relatively simple installation. Very high fuel-flow rates are required for adequate cooling; consequently, the fuel-cooling system should be considered only for afterburning engines. For the particular fuel-cooling system analyzed, up to 10 minutes of nonafterburning cruising operation could be provided with heat rejection to the main fuel tanks during initial portions of the flight. The most promising application

of the fuel-cooling system is the guided missile at very high flight Mach number in which continuous afterburner operation is required; thus, heat rejection to the fuel tanks is avoided. A regenerative liquid-cooling system, which rejects heat to compressor-discharge air, can be provided for nonafterburning engines as well as for afterburning engines; if suitable fluids can be found, this system provides a "package" installation entirely contained within the engine and capable of operation without external systems in the aircraft. 2452

INTRODUCTION

The projected development of high-performance turbojet-powered military aircraft, such as the supersonic interceptor and the guided missile, introduces new engine-performance requirements that, in turn, influence design of the basic engine components. The installed thrust capacity of a supersonic interceptor, for example, tends to become very large as the specified combat speed, altitude, and maneuverability are increased relative to currently acceptable values. Provision for adequate specific engine thrust and minimum specific weight to meet the new engine-performance requirements depends upon evolution of the basic compressor and turbine components in the direction of increased air-handling capacity, increased blade tip speed to reduce compressor length and hence weight, and increased turbine-inlet gas temperature.

The combination of increased specific mass flow and blade tip speed to minimize engine weight results in severe stresses in the rotating parts that are beyond the capacity of the best available high-temperature turbine-blade alloys at the high gas temperatures desired. Suitable turbine-cooling systems are therefore required in future high-performance engines to control blade temperature and thereby increase the effective strength of the blade alloy. The performance of aircraft at supersonic flight speed is relatively insensitive to compressor pressure ratio; therefore, compressor weight can be minimized by the utilization of a minimum number of transonic or supersonic stages having increased air-handling capacity. High blade tip speeds are necessary, however, to achieve a high pressure ratio per stage, and the turbine must be effectively cooled with air or liquids to permit a simultaneous increase in gas temperature and blade stress level.

At supersonic flight speeds, the problems encountered in cooling the turbine blades become increasingly severe, and the turbine-cooling

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limitations probably establish the eventual limit of attainable turbojet-engine performance without afterburning at supersonic speed and high altitude. In the case of liquid cooling, the turbine-cooling system must be capable of rejecting the direct cooling load to whatever heat-rejection mediums are available in the engine installation. With the wide variety of cooling arrangements which must be considered, it is evident that application of a turbine-cooling system will influence the engine installation and possibly the aircraft configuration with regard to disposition of fuel tanks, engine nacelles, inlets, ducts, and fuel plumbing systems.

Analytical and experimental research has already been conducted on the application of air-cooled turbine rotors to typical production turbojet engines so that substitution of noncritical low-alloy-steel turbine blades and disks may be permitted (references 1 to 6). Analyses have been made to determine methods of calculating liquid-cooled turbine-blade temperatures (reference 7), and experimental heat-transfer data have been obtained with a liquid-cooled turbine (reference 8) at current gas temperature and stress levels. An analysis of the cooling effectiveness of several liquid-coolant circulation systems for turbine blades is also presented in reference 9. However, the currently available experimental turbine data and full-scale engine experience are inadequate for the over-all comparison of air- and liquid-cooling systems required in future high-performance engines to permit a simultaneous increase in turbine-inlet temperature, specific mass flow, and blade tip speed.

An analytical investigation was conducted at the NACA Lewis laboratory to evaluate the general cooling characteristics of various turbine air- and liquid-cooling systems and to compare the applicability of air- and liquid-cooled turbojet engines. Since the evaluation of turbine heat-transfer characteristics requires knowledge of the actual physical dimensions of the cooled components in addition to the conventional procedures for thermodynamic analysis, it becomes necessary to work with one hypothetical engine and fixed aircraft design specifications to obtain quantitative comparisons from which a subsequent general comparison of air- and liquid-cooling systems can be made. It is essential, however, that the design and operating conditions selected as a basis for comparison be within the range of practical interest. The operating conditions imposed in this analytical investigation were therefore based on an assumed supersonic interceptor flight plan over a range of flight conditions up to a Mach number of 2.5 at an altitude of 50,000 feet. The cooled-turbine design analysis was made for an assumed basic engine of a type and size appropriate for the gross weight and power loading of a supersonic interceptor aircraft. The basic engine design specifications, which represent a considerable advance in some respects over

turbojet engines currently in production, include a sea-level specific mass flow of 23.6 pounds per second per square foot of compressor frontal area, a turbine-inlet temperature of 2040° F, a sea-level compressor pressure ratio of 6.0, a blade tip speed of 1500 feet per second, a turbine diameter of 35.1 inches, and a turbine hub-tip ratio of 0.732. In the liquid-cooling-system analysis, three ultimate heat-rejection mediums were considered: ram air, afterburner fuel, and engine working fluid.

The various liquid-cooling systems are considered in this report. The characteristics of air-cooling systems and their comparison with liquid-cooling systems are presented in reference 10.

Specific objectives of the liquid-cooling systems analyzed were:

(1) To provide a comparison of several possible systems under specified engine and flight conditions,

(2) To determine which flight condition represented the most critical cooling requirement,

(3) To determine the limitations imposed on rotor-blade cooling effectiveness by the available heat-rejection mediums,

(4) To evaluate modifications that could be made to the cooling cycle to avoid limitations imposed by the heat-rejection mediums,

(5) To evaluate thermodynamic possibilities for completely self-contained regenerative liquid-cooling systems, independent of engine installation or auxiliary systems and rejecting heat to the main-engine working fluid at compressor-discharge conditions.

In this analysis no consideration was given to cooling the stator blades.

Engineering data were not available to permit a complete analysis of installed weight and performance of the cooled engine and various systems for heat rejection. The comparison of liquid-cooling systems could therefore be based only upon an evaluation of over-all cooling effectiveness; design, installation, and operating problems; degree of mechanical complication; probable weight penalties encountered; and capacity of the cooling system to accommodate increasingly severe design specifications in future applications. In this manner, a perspective is revealed that indicates the extent to which the turbine constitutes a basic limitation in the applicability of turbojet engines of improved design and the turbine-blade-cooling systems that offer the most promise for research and development. As a part of this evaluation, results of the analysis are given in the form of figures and tables which indicate the influence of important design variables and which constitute the basis for the engineering evaluation.

DESCRIPTION OF LIQUID-COOLING SYSTEMS INVESTIGATED

Coolant-circulation systems within turbine rotor. - Liquid-cooling systems for turbine-rotor blades are discussed in reference 9 from the standpoint of the heat transferred to the coolant and the natural-convection pumping forces developed in the fluid as a result of the intense centrifugal field in the rotor. The reference indicates that the thermosiphon pumping action induced far exceeds the pressure drop
2452 in the coolant passages so that one of two possible courses of action is necessary to control the rate of coolant flow, the blade temperature, and the coolant temperature rise. The first is to throttle the flow of the coolant in the coolant-passage outlet resulting in a straight-through circuit. The second is to provide a return path from the passage for the fluid leaving the turbine rotor to the passage for the fluid entering the turbine rotor, thus allowing the coolant to recirculate through the blade coolant passages several times before leaving the turbine rotor. This method of dissipating the excess pumping force by overcoming the increased pressure drop resulting from the higher recirculation rates through the coolant passages is known as the loop circuit. Both methods of controlling the coolant-flow rate in the rotor, straight-through and loop circuits, are considered herein.

Water-cooling system. - The simple water-cooling system with heat rejection to ram air is illustrated schematically in figure 1. This system is very similar to those systems used in the past for liquid-cooled reciprocating aircraft engines and represents a familiar basis for comparison with other cooling systems. In the water-cooling system, the coolant is passed through the turbine disks and blades in a closed circuit that includes an external radiator located in a separate duct which accommodates the flow of ram air taken aboard the aircraft for heat rejection. The water enters the turbine rotor at a low temperature and, after passing through the blades with a relatively small temperature rise, is pumped through the radiator where it gives up heat to the ram air. The entire water system must be pressurized to the level required to avoid local boiling in the turbine and the external parts of the system. The radiator is located as close to the engine as possible so as to minimize the total weight of fluid contained within the system. In the installation assumed for this analysis, the radiator and its inlet, controls, and jet nozzle were assumed to be located in an independent propulsive duct within the engine nacelle for thrust recovery. Several alternate arrangements not analyzed in this report should be noted. One arrangement utilizes a fuselage installation in which a large amount of air is available from boundary-layer-removal slots placed adjacent to the main-engine air inlets for efficient ram recovery at high flight speed. In this case, the radiator duct would not necessarily be designed for thrust recovery but would utilize low-energy air originally taken aboard the aircraft for another purpose. A second arrangement of the water-cooling system utilizes the afterburner fuel as the ultimate heat-rejection medium.

Fuel-cooling system. - Another liquid-cooling system, and one that can be applied with afterburning engines, utilizes afterburner fuel as the heat-rejection medium. A fuel-cooling system with heat rejection to the afterburner fuel is illustrated schematically in figure 2. In this system, the afterburner fuel is taken from the fuel tank and passed directly through the turbine disks and blades in an open circuit and then pumped to the afterburner fuel-injection nozzles. Although a liquid hydrocarbon fuel does not have as desirable heat-transfer characteristics as water, it is readily available on the aircraft in sufficient quantity to avoid the necessity for an external radiator to reject heat to ram air. The turbine-cooling system can therefore be made relatively independent of flight and operating conditions provided that the deficiencies of fuel as a coolant can be overcome through careful design of the blade coolant passages. Since the coolant is already carried by the aircraft, there is no appreciable additional weight factor to be considered in the design. The high pressures required to avoid vaporization of the fuel within the system can be obtained by thermal pumping action in the turbine, and these high pressures are desirable for the injection of the preheated fuel into the afterburner fuel nozzles.

An important aspect of the fuel-cooling system is the provision for nonafterburning engine operation in which the required amount of fuel for cooling is recirculated between the fuel tank and the turbine. The heated fuel emerging from the turbine is passed back into the fuel tank where it is cooled temporarily when mixed with the large amount of fuel stored in the tank. In this manner, the turbine cooling load, during periods of nonafterburning engine operation, is stored in the main fuel tank and later rejected through the afterburner fuel nozzles during normal afterburning operation. It is obvious that this process can continue for only a limited time, determined by the initial fuel load and temperature, and the permissible pressurization of the fuel tanks to prevent evaporation losses of the fuel. This suggests that only aircraft configurations suitable for fuselage fuel tank installation can be considered for this application because of the necessity for stable pressure vessels that cannot readily be incorporated within a thin wing structure.

In all cases considered in this phase of the analysis, the heat loss in the turbine blades is utilized to preheat the fuel for combustion. Only afterburner fuel was used for turbine cooling because the primary fuel flow is generally too small for effective cooling and is ordinarily used in modern engine installations to cool the lubricating oil.

Regenerative refrigeration system. - If neither ram air nor afterburner fuel were considered desirable for use in a liquid-cooled engine installation, it would become necessary to provide a self-contained regenerative system that is capable of rejecting the turbine-cooling

load to the main-engine working fluid. Because of the detrimental effect upon compressor performance, rejection of heat ahead of the compressor inlet is very undesirable. It is therefore necessary for heat rejection to occur at compressor discharge just prior to combustion.

A regenerative liquid-cooling system with heat rejection to the engine working fluid at compressor discharge is illustrated in figure 3. The distinguishing feature of this system as compared with those previously discussed is that the energy level of the coolant must be raised sufficiently by a heat-pump cycle to allow heat rejection to the hot air behind the main-engine compressor. In order to minimize the coolant-flow rate and achieve the maximum possible heat-transfer rates, the system operates on a vapor cycle that utilizes the latent heat of vaporization of the coolant in both the heating and cooling portions of the cycle. In the system illustrated in figure 3, the coolant is introduced as a liquid into the turbine, and after circulating through the blade coolant passages as a liquid, a portion of the heated fluid is allowed to flash to a vapor within the turbine rotor. The portion of the heated fluid which remains in liquid form is recirculated through the turbine rotor. The vapor, on entering the refrigeration cycle, is first compressed in a small auxiliary engine-driven compressor so that the saturation temperature is higher than the main-engine compressor-discharge temperature and then passed into a condenser located in the main stream at compressor discharge. The heat transferred in the turbine, as well as the power used to raise the energy level of the vapor, is rejected in the condenser. The condensed liquid passes next through an expansion valve, as in standard refrigeration processes, where most of the coolant is subcooled to the specified turbine-coolant-inlet temperature at the expense of some residual flash vapor that must be recirculated constantly through the system. The regenerative system is very similar to conventional refrigeration apparatus except that the turbine is used as an evaporator and the condenser is located inside the engine. This system is best suited to chemically stable, high-boiling-point liquids that permit high turbine-coolant temperatures without excessive internal pressures.

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Operation of a refrigeration cycle, such as that illustrated in figure 3, is greatly influenced by the temperature ratio that must be maintained between the turbine-coolant-inlet temperature and the condensing-liquid temperature. If this temperature ratio is sufficiently low, the thermal pumping action of the turbine may be used instead of a separate engine-driven compressor for the heat-pump cycle, thus considerable mechanical simplification is achieved.

ANALYTICAL METHODS FOR EVALUATING LIQUID-COOLING SYSTEMS

The general analytical procedure followed is similar to that presented in references 11 and 12, where the turbine configuration and engine operating conditions over a range of altitude and flight speed are used to define the most essential factors in the evaluation: heat-transfer rates, fluid temperatures, coolant flow available or required, and blade temperature. At all operating conditions, the blade temperature must be held within a safe operating limit which is determined by the stress level in the engine and the strength characteristics of the materials. In general, liquids are more effective than air as coolants and, even with moderate flow rates, usually reduce the blade temperature considerably more than would be required on the basis of material strength alone. The rate of liquid circulation through the turbine may therefore be varied independently within limits established by factors other than blade temperature.

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Design Criteria

A turbine-cooling system that can accommodate the specified turbine-inlet temperature and engine mass flow with the coolant-flow rates available is presumed to be adequate and fulfills the first essential criterion of evaluation. The design criteria used in evaluating the cooling systems vary as previously indicated, a fact which results in different types of calculations. As a result of the high effectiveness of liquids as coolants, the blade temperature is relatively easy to control and the analysis is mainly concerned with the heat-rejection rates, temperatures, and pressures in the system.

In the water-cooling system, the radiator is the principal problem because radiator size is influenced by the magnitude of the heat-rejection rate. In order to reduce the size and weight of the radiator, it is desirable to maintain a large temperature difference between the turbine-cooling water and ram air at the face of the radiator core. This requires high pressures within the system, but these pressures are limited by structural problems in the radiator and in the plumbing, where light-weight designs are required. The relatively low coolant temperatures necessary to avoid excessive pressures in the water-cooling system result in low blade temperatures with high heat rejection from the gas to the turbine blade. If unstable local internal boiling is to be avoided, the temperature rise of the coolant through the turbine is pressure limited; therefore, large circulation rates must also be maintained. To evaluate this system it is first necessary to relate the blade-profile heat-transfer coefficients to the engine gas flow and to determine the coolant-passage heat-transfer coefficients for a range of coolant flows. Then, with consideration of the permissible coolant temperature and temperature rise, as well as the blade temperature level

desired, the heat-rejection rate for the radiator is determined. This is done over the range of flight conditions to determine which condition represents the most critical situation since the heat-rejection rate varies primarily with the engine mass flow, whereas the temperature of the ram air to which heat is being rejected varies in a different manner with flight Mach number and altitude.

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The design criteria for the fuel-cooling system differ from those of the water-cooling system in that the cooling effectiveness of the fuel is less than that of water. The low heat capacity of fuel (about one-half that of water) results in a larger coolant temperature rise within the turbine and higher blade temperatures. These conditions lead to chemical instability of the fuel and provide a serious limitation to its use as a coolant. The combination of high pressure and temperature in the blade coolant passage is believed to be sufficient to cause "cracking" of the fuel, which may upset combustion stability in the afterburner and lead to cumulative formation of deposits and sludge in the blade coolant passages. To evaluate the fuel-coolant system, the blade profile and coolant-passage heat-transfer coefficients must be evaluated and the heat-transfer rate determined with consideration of the permissible fuel temperatures. The fuel temperature coming from the tanks is low and probably independent of the flight conditions, but the fuel-flow rate to the afterburner is directly proportional to the engine mass flow, whereas the heat-rejection rate for the turbine varies exponentially with the gas flow. The analysis must therefore be made over a range of flight Mach numbers and altitudes to determine which condition is most critical with respect to fuel temperature rise and to determine whether the cooling system is adequate at all points in the interceptor mission. For nonafterburning engine operation, where the fuel is recirculated to the tanks, an analysis of fuel-tank temperature rise and pressure as a function of loiter time must be made to determine the duration of flight without severe evaporation losses or the degree of tank pressurization required to meet a specified loiter time.

The regenerative liquid-cooling system is characterized by the highest coolant temperatures where the design criteria are determined by the latent heat characteristics of the coolant, saturation pressures of the vapor, and the temperature ratio between the condenser and evaporator, which is the turbine rotor in this case. A major problem is selection of a fluid that has desirable heat-transfer properties, chemical stability, and vapor pressure characteristics suitable for the temperature level and temperature ratio over which the refrigeration cycle must operate. The temperature level throughout the refrigeration cycle is to a large extent established by the temperature of the main-engine compressor-discharge air. In general, high blade- and coolant-temperature level contribute to minimum external work in the heat-pump portion of the refrigeration cycle and minimum heat-rejection rate in the condenser. In order to evaluate the regenerative refrigeration

system it is first necessary to calculate the turbine heat-rejection rates and coolant temperatures as previously indicated for the other liquid-cooling systems. Criteria are then established for the evaporation of a portion of the coolant and for the compression of this vapor to a state that will permit rejection of heat to the engine working fluid. The heat-rejection capacity of the condenser, which must also be determined, is influenced by two factors, the heat rejection to the coolant and the work required to raise the energy level of the coolant 2452 to that of the condenser. Another significant item that must be evaluated in the refrigeration cycle, along with the temperatures and pressures of the refrigerant, is the quantity of residual flash vapor that must be recirculated continually to accomplish refrigeration in the expansion process.

Equations for Heat Rejection, Prevalent Blade Temperature, and Coolant Temperature

The heat-transfer rate from the hot gas to the liquid coolant for the straight-through, as well as the loop, circuit is expressed by each of the following equations:

$$Q = H_o S_o \left(T_{g,e} - T_{B,m,o}\right) \tag{1}$$

$$Q = k_B \frac{S_o}{\Delta y} \left(T_{B,m,o} - T_{B,m}\right) \tag{2}$$

$$Q = H_{l,i}\, S_i \left(T_{B,m} - \frac{T_1 + T_2}{2}\right) \tag{3}$$

$$Q = w_l\, c \left(T_2 - T_1\right) \tag{4}$$

All symbols are defined in appendix A. As in reference 9, a constant thickness of blade material Δy was considered for heat transfer, and it was assumed that there was no heat pickup in the disk by the coolant. Consideration will first be given to the straight-through circuit. The coolant-flow rate w_l, effective gas temperature $T_{g,e}$, coolant-inlet temperature T_1, gas-to-blade heat-transfer coefficient H_o, and blade-to-coolant heat-transfer coefficient $H_{l,i}$ are specified or are determined by separate calculation. A simultaneous solution of the foregoing equations provides the rate of heat rejection to the coolant Q, the mean or prevalent blade temperature on the blade outer surface $T_{B,m,o}$, the mean or prevalent blade temperature near the coolant

passages $T_{B,m}$, and the coolant-outlet temperature T_2. For the loop circuit, $H_{l,i}$ must be determined from separate calculations as a function of temperature rise and coolant flow (reference 9). The value of $H_{l,i}$ obtained must simultaneously satisfy equations (1) to (4) for the assumed coolant temperature rise $(T_2 - T_1)$ and the coolant flow w_l. These equations may then be solved for the rate of heat rejection to the coolant Q, the prevalent blade temperature on the blade outer surface $T_{B,m,o}$, and the prevalent blade temperature near the coolant passages $T_{B,m}$.

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Gas-To-Blade Heat-Transfer Coefficient

A theoretical method of determining an average gas-to-blade heat-transfer coefficient has been developed in reference 13. The theory shows good agreement with experimental results obtained on a water-cooled aluminum turbine and in static cascades of air-cooled blades (references 14 to 16). These theoretical methods were used to determine the average gas-to-blade coefficients in this analysis. The expression for the gas-to-blade coefficient in dimensionless form is

$$Nu_{g,B}/(Pr_{g,B})^{\frac{1}{3}} = \bar{F}(Re_{g,B})^{Z} \tag{5}$$

where

$$Nu_{g,B} = H_o \frac{l_o}{\pi} \Big/ k_{g,B}$$

$$Pr_{g,B} = c_{p,g,B}\, \mu_{g,B}\, g/k_{g,B}$$

$$Re_{g,B} = \rho_{g,B}\, W_{g,m} \frac{l_o}{\pi} \Big/ \mu_{g,B}$$

where the values of $\bar{F}$ and Z may be obtained from dimensionless charts in reference 13. Fluid properties are based upon blade-wall temperature.

Blade-To-Coolant Heat-Transfer Coefficient

Straight-through circuit. - Results of an experimental investigation on a water-cooled turbine (reference 8), where the flow of liquid through the coolant holes in the blades was laminar, have indicated that heat-transfer coefficients for liquids in pipes can be used with little error

to determine the inside heat-transfer coefficients for the turbine blades. It was assumed, on the basis of the tests in reference 8, that formulas for heat transfer in pipes for turbulent flow could be used to calculate the coefficients in the blades.

The following equation from page 168 of reference 17 was used. It is applicable to fluids having viscosities of not more than twice that of water and for Reynolds numbers exceeding 2100. 2452

$$\frac{H_{l,i}D}{k_l} = 0.023 \left(\frac{w_l D}{A g \mu_l}\right)^{0.8} \left(\frac{c \mu_l g}{k_l}\right)^{0.4} \tag{6}$$

where D is the coolant-passage diameter and A is the cross-sectional area of the coolant passages through which the quantity of coolant w_l flows. Fluid properties are based on coolant bulk temperature.

Loop circuit. - In the case of the loop circuit, the coolant-flow rate giving rise to the heat-transfer coefficient in the holes can be several times as great as the flow rate w_l into the rotor, depending on the number of times the coolant is recirculated through the passages before leaving the turbine rotor. The number of times the fluid is recirculated is determined by the thermosiphon pumping action generated, which in turn is dependent on the temperature rise of the coolant in the passages.

The heat-transfer coefficients and the amount of fluid flowing through the passages due to the recirculation for loop circuits have been worked out and charts are presented in reference 9 from which they can be determined.

Determination of Water-Radiator Characteristics

The characteristics of the radiator used in the water-cooling system (fig. 1) were determined from references 18 to 20. The sizes of the radiators were determined from curves in reference 18, with the calculated heat-rejection rates and water-coolant-flow rates known. The weight of the radiators was then determined from data of reference 19 and the previously determined sizes. The pressure drop across the air side of the radiator, the drag resulting from this pressure drop, and the inlet diffusion losses were then calculated from data of reference 20.

Fuel Temperature in Tank with Fuel Cooling (Nonafterburning Operation)

When the flight plan requires operation without the afterburner and when fuel cooling of the blades is used, an alternate heat receiver other than the afterburner must be provided. During the early phase of a flight a large quantity of fuel is carried aboard the aircraft, and the possibility of using this fuel load as a temporary heat-storing medium was investigated. The use of a system of this type (see fig. 2) is limited by the allowable temperature rise of the fuel in the tanks, which is dependent on the rate at which heat is rejected to the fuel, the weight of fuel at the beginning of the heat-rejection process, and the rate at which fuel is taken from the tanks by the operating engines. The following equation, which is applicable to any method of heat rejection to fuel, is derived in appendix B and was used to calculate the temperature rise of the fuel in the tank during periods of nonafterburning operation.

$$T_f - T_{f,0} = -\left[\frac{Q}{c(f)w_C}\right] \ln\left(\frac{w'_{f,0} - fw_C\,\tau}{w'_{f,0}}\right) \tag{7}$$

where

T_f temperature of fuel in tank

$T_{f,0}$ initial temperature of fuel in tank

w_C engine-compressor-air weight flow

$w'_{f,0}$ initial weight of fuel in tank

The amount of fuel in the tank after τ seconds is obtained from the expression $(w'_{f,0} - fw_C\,\tau)$. The quantity of heat rejected Q used in the equation is determined from heat-balance equations presented previously.

Regenerative Cycle Calculations

In the regenerative liquid-cooling system, the amount of liquid evaporation required, the work of compression on the liquid evaporated, and the percentage of liquid which evaporates during the throttling process downstream of the condenser (see fig. 3) are factors which are necessary in the evaluation of the system. From the temperatures required in the turbine blades and the compressor-discharge temperature, the liquid most favorable for the system can be selected from the saturation properties of the liquids.

The amount of liquid evaporation required depends on the rate of heat rejection in the turbine blades and is readily calculated from the following equation:

$$w_v = \frac{Q}{h_{l,v}} \tag{8}$$

where $h_{l,v}$ is the heat of vaporization of the liquid. The rate of heat rejection was calculated by the equation for heat transfer from combustion gas to outer blade surface (equation 1). 2452

The work needed to compress the vapor leaving the turbine rotor to the pressure required in the condenser is calculated according to conventional methods which require knowledge of auxiliary compressor pressure ratio, initial vapor temperature, and efficiency of the auxiliary compressor.

The amount of liquid which evaporates during the throttling process (a constant-enthalpy process) and the amount of vapor which is recompressed and taken back to the condenser can be determined from the following equation:

$$h_{l,3} = Xh_{l,4} + (1 - X)\, h_{v,4} \tag{9}$$

where

X	fraction of liquid after throttling
$1-X$	fraction of vapor after throttling
$h_{l,3}$	enthalpy of liquid before throttling
$h_{l,4}$	enthalpy of liquid after throttling
$h_{v,4}$	enthalpy of vapor after throttling

The enthalpy $h_{l,3}$ is determined at the temperature downstream of the condenser. The liquid is throttled to the pressure at which evaporation takes place in the turbine. At this pressure both liquid and vapor are at the saturation temperature after throttling. Enthalpies $h_{l,4}$ and $h_{v,4}$ can then be obtained from saturation tables at the evaporation pressure in the turbine.

AIRCRAFT AND ENGINE DESIGN OPERATING CONDITIONS

Assumed Interceptor Flight Plan

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A supersonic interceptor aircraft mission was chosen for this analysis. Flight conditions are listed in table I. It was specified that the aircraft take off and accelerate to a Mach number of 0.8 at sea level, climb to 35,000 feet at a Mach number of 0.8, accelerate to its combat Mach number of 1.8 at a constant altitude of 35,000 feet, and finally climb to 50,000 feet at the combat Mach number of 1.8. It was further specified that the combat portion of the mission consist of a constant 2g maneuver at an altitude of 50,000 feet and a Mach number of 1.8 without loss of speed or altitude for the duration of the available fuel supply. The thrust requirement for the combat maneuver was sufficient to permit level flight at a Mach number of 2.5 and an altitude of 50,000 feet. If loiter time was required during the mission prior to combat, it was assumed that this was carried out at an altitude of 35,000 feet and a Mach number of 0.8.

Assumed Engine Design Operating Conditions

The principal engine design operating conditions assumed for the analysis, which must be specified before the engine size can be determined, were turbine-inlet temperature of 2040° F at all points in the mission, sea-level static compressor pressure ratio of 6.0, and afterburner-inlet temperature of 3040° F at all points in the mission.

Aircraft and Engine Configuration

The assumed aircraft was a straight-wing supersonic configuration powered by two afterburning turbojet engines, each located in a nacelle at the wing tips. The performance was based on representative data. The assumed gross weight of the aircraft at take-off was approximately 28,000 pounds. The installed engine thrust capacity for the aircraft was determined by the assumed 2g combat maneuver at maximum speed and altitude. The sea-level installed weight of the engines was approximately 26 percent of the aircraft gross weight, based on an assumed unaugmented specific engine weight of 0.28 pound per pound thrust. The aircraft structure was assumed to be 30 percent of the gross weight, the pay load, 10.7 percent of the gross weight, and the fuel and tanks composed the remainder of the disposable load. A representative weight distribution at take-off for such an aircraft is as follows: pay load, 3000 pounds; structure, 8400 pounds; fuel tanks, 870 pounds; fuel, 8500 pounds; and both engines, a total of 7230 pounds. The installed engine weight is seen to be almost equal to the total fuel load, thus

emphasizing the critical importance of maintaining a minimum specific engine weight. An increase of 50 percent in specific engine weight in this example would result in approximately a 39-percent decrease in fuel capacity.

Since the installed engine thrust capacity is determined by the combat condition, the assumed aircraft is overpowered at all other conditions, and if afterburning operation is assumed throughout the mission, the thrust-to-gross-weight ratio is approximately 1 at sea level. 2452
Combat speed and altitude are reached in less than four minutes, and with the limited available fuel supply, the total flight time is approximately 15 minutes; this total does not include loitering time. If loitering time is added, the total flight time is naturally increased but combat time would be decreased. It was assumed that loiter time at a Mach number of 0.8 and an altitude of 35,000 feet could be accomplished simply through nonafterburning engine operation although the turbine-inlet temperature would probably have to be reduced, as well, from the specified value of 2040° F to permit sustained operation at this low flight speed.

The thrust requirements for the assumed interceptor aircraft are approximately met with an engine designed for turbine-inlet temperature of 2040° F, an afterburner temperature of 3040° F, a sea-level static compressor pressure ratio of 6.0, and a mass flow of 158 pounds per second. In the interest of minimum specific engine weight, compressor tip speed was assumed to be 1500 feet per second, thereby minimizing the number of compressor stages. The specific mass flow of the compressor was 23.6 pounds per second per square foot. The assumption of a basic engine with known dimensions and compressor characteristics is necessary to reflect adequately the influence of the off-design operating characteristics of the engine components on the heat-transfer analysis at high altitude and supersonic flight speed and does not affect the comparison of the various cooling systems.

The important engine operating conditions, component performance data and flow rates of air and fuel at significant points in the flight plan, are summarized in table II. These values, obtained from general cycle analysis for the main points of interest in the flight plan, provide the basic numbers for the cooling analysis. The tabulated values show that the engine mass flow, which is an important factor in determining the heat-transfer rates in the turbine, may exceed the sea-level static condition over portions of the flight plan. This indicates that critical altitude cooling problems may be encountered. Also, the ram temperature reaches 186° F and the compressor-discharge temperature reaches 604° F at combat conditions. At the maximum Mach number of 2.5 encountered in level flight at 50,000 feet, the ram temperature was 423° F and the compressor-discharge temperature was 802° F. These excessive temperatures further complicate the heat-rejection problem.

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Turbine Design

A summary of the turbine specifications is given in table I. The turbine of the basic engine was limited to a single stage having a tip speed of 1500 feet per second and a tip diameter of 35.1 inches. Fifty-eight constant cross-section blades having a span of 4.72 inches and a chord of 2.27 inches were provided. The turbine-blade profile used in this analysis is shown in figure 4. This blade was not designed specifically for the turbine used in this analysis; however, calculations showed that such a blade would approximately fulfill the design conditions on the basis of the required velocity diagram for driving the compressor. The blade profile is the same as profile 2 in reference 21. This profile is adaptable to either liquid or air cooling, and coolant passages can be placed reasonably near the leading and trailing edges which normally are the hot spots in a cooled blade. The hub-tip ratio of the turbine was 0.732, and this, together with the tip speed of 1500 feet per second, indicates the severe mechanical design requirements that must be met with the application of transonic and supersonic compressor stages.

The spanwise distribution of centrifugal stress in the untapered turbine blade of this analysis is shown in figure 5. Provision for a tapered blade is unlikely in liquid-cooled blades because coolant passages would probably be placed so as to provide a minimum wall thickness so that temperature gradients in the leading and trailing edges could be minimized. Figure 5 may be used in conjunction with stress-to-rupture data or yield-strength data of the material under consideration to determine the allowable blade temperature. Yield-strength data are generally applicable for high-temperature alloys in a temperature range below 1000° F. For the blade profile under analysis, figure 5 shows that the centrifugal stress at the blade root is 60,000 pounds per square inch.

CALCULATION PROCEDURE

In order to evaluate and compare the liquid-cooling systems, the methods of analysis outlined previously were applied for the basic engine and flight operating conditions specified for the typical interceptor aircraft. The blade design of reference 21 was considered as approximately fulfilling the design conditions of this analysis. Gas-to-blade heat-transfer coefficients were calculated from equation (5), which is expressed as follows:

$$Nu_{g,B}/(Pr_{g,B})^{0.333} = 0.14\ Re_{g,B}^{0.662} \qquad (10)$$

For the conditions considered in the present report, the values of $\overline{F}$ and Z (equation (5)) would probably vary slightly from the values given in equation (10) if a rigorous design were considered. The effect of this variation on the over-all results and conclusions drawn in this analysis would be negligible, however, and a rigorous blade design procedure is therefore not warranted. The gas-to-blade heat-transfer coefficient was calculated from equation (10) in a manner similar to that employed in reference 21. 2452

Blade-to-coolant heat-transfer coefficients for the straight-through circuit were calculated from equation (6). Calculations were made for a coolant-flow ratio of 0.03 as part of the cooling-system analysis and for the complete range of coolant flows given in table I to provide a comparison of coolant-passage heat-transfer coefficients. A cross section of the liquid-cooled blade used in this analysis is shown in figure 4. Each blade was assumed to have six coolant passages, three of which carried the coolant radially outward while the other three returned the coolant radially inward. The outward flow and return flow passages were interconnected with cross-over passages in the blade tip. Coolant passages of 0.125-inch diameter were assumed. For this size passage and the coolant-flow quantities considered (table I), it was determined that the flow would be in the turbulent regime and equation (6) therefore applicable.

Blade-to-coolant heat-transfer coefficients for the loop circuit were also calculated for the blade-coolant-passage configuration shown in figure 4. For each value of coolant flow entering the turbine w_l and coolant temperature rise $(T_2 - T_1)$, a Grashof number was calculated. The investigated range of coolant flows entering the turbine is given in table I. From the flow rate into the turbine, the parameter $w_l/D\mu_l g\ N$ was calculated. Then, from liquid properties and nondimensional charts in reference 9, the heat-transfer coefficient and the quantity of fluid passing through the blade passages was obtained. The true heat-transfer coefficient, however, is one which must simultaneously satisfy the loop-circuit equations of reference 9 as well as the heat-balance equations (equations (1) through (4)). Consequently, for each coolant weight flow into the turbine and each desired gas temperature, there is only one permissible coolant temperature rise. The magnitude of this temperature rise is a factor in the evaluation of the loop system.

The comparison of water- and fuel-cooling systems (figs. 1 and 2) was based upon calculations of blade temperature and heat-rejection rates over the entire mission. The coolant-flow ratio for the fuel-cooled turbine as well as for the water-cooled turbine was set at 0.03 which was approximately equal to the afterburner fuel-air ratio. A turbine coolant-inlet temperature of 150° F was set for both turbines. The critical point in the mission was then determined with respect to

heat rejection from the blades to the coolant. Consideration of the coolant temperatures and pressures leaving the turbine then indicates the severity of the ultimate heat rejection to ram air for the water-cooling system and permits selection of a critical flight condition for analysis of the water-cooling radiator. The radiator was checked at the critical flight conditions and estimates were made of the size, weight, drag, and installation problems.

2452 For the nonafterburning case where fuel is used as blade coolant and returned to the fuel tank (a condition which arises when the aircraft is loitering at 35,000 ft before interception), the rate at which heat was removed from the blades and rejected to the fuel remaining in the tank was determined for the same mean outside-surface blade temperature as was obtained in the previous calculation for heat rejection to afterburner fuel. The temperature rise in the tank and the quantity of fuel remaining in the tank during the time of loitering or cruising were then calculated. It was assumed that during the climb to cruise altitude and speed, the afterburner would be operating and the tank-fuel depletion rates used were consequently those obtained with afterburner operation. Because of the excess cruising power at a flight Mach number of 0.8 and an altitude of 35,000 feet, the tank heat-rejection rates were calculated for both the normal 2040° F turbine-inlet temperature and a reduced turbine-inlet temperature of 1620° F that resulted in half the rate of heat rejection. As previously mentioned, reduced inlet temperature is probably required to permit reduced thrust cruising at subsonic speeds. The schedule of fuel temperature rise with duration of flight was then used to determine the permissible loiter time. The limiting tank temperature was taken as 150° F because, at high altitudes, excessive pressures might be encountered in preventing evaporation of the fuel if higher fuel tank temperature were permitted.

The regenerative liquid-cooling system was evaluated with a heat-rejection rate determined on the basis of a mean outside-surface blade temperature of 1060° F. Although the temperatures and pressures encountered with the water-cooling system make its use as a refrigerating vapor impractical, it was used as an illustration because it has known heat-transfer properties and complete thermodynamic data available in the superheat region. The peak compressor-discharge temperature was evaluated from engine-cycle calculations which established the temperature and pressure in the refrigeration cycle at the condenser outlet. The refrigeration capacity over this temperature and pressure ratio was then evaluated for only one flight condition to illustrate the relative magnitude of the energy terms and the amount of shaft power extraction needed to drive the auxiliary compressor. An auxiliary compressor efficiency of 0.80 was assumed for this analysis.

RESULTS AND DISCUSSION

Water and Fuel Coolant-Passage Heat-Transfer Coefficients

A basis for comparison of liquid coolants is the coolant-passage heat-transfer coefficient obtainable over a range of coolant flows. This comparison emphasizes the differences that occur because of the physical properties of the fluid, the method of circulation through the coolant passages, and the influence of coolant-passage design. 2452

Straight-through system. - Coolant-passage heat-transfer coefficients obtainable with water and fuel in a straight-through system for the blade used in this analysis are given in figure 6. Coefficients are presented for a range of coolant flows of 0.03 pound per second per blade to 0.10 pound per second per blade. The latter flow is equivalent to 3.67 percent of the engine mass flow at sea-level static conditions. The heat-transfer coefficients obtainable with water are shown to be much greater than those obtained with fuel, a fact largely due to differences in the physical properties of the two fluids, such as viscosity, specific heat, and thermal conductivity. For a typical coolant flow of 0.082 pound per second per blade, which corresponds to a coolant-flow ratio of 0.03 at sea-level static conditions, the heat-transfer coefficient with fuel is 740 Btu/(hr)(sq ft)($^{\circ}$F), whereas the heat-transfer coefficient with water is 2860 Btu/(hr)(sq ft)($^{\circ}$F). Thus, for the same weight flow of coolant, the heat-transfer coefficient obtainable with fuel is about one-fourth of that obtainable with water.

The high heat-transfer coefficients which may be obtained with water pose the problem of overcooling the blade, thus resulting in unnecessary losses and possibly severe thermal stresses. Two remedies are suggested: one is to arbitrarily reduce the coolant-flow rate through the turbine; the second is to increase the coolant-passage diameter while simultaneously reducing the number of passages. Both of these methods must be applied cautiously, however. For example, arbitrary reduction of the coolant-flow rate results in a higher coolant temperature rise. Higher pressures in the coolant discharge are then required to avoid local boiling conditions which might interfere with uniform coolant distribution in the blade coolant passages and cause uncontrolled hot spots in the blade metal. Reduction in the number of coolant passages must also be considered from the standpoint of probable increased thermal stresses because of the longer conduction paths established in the blades. Thus it is apparent that a proper balance must be achieved between several limiting conditions in order to provide a desirable water-cooled straight-through-circuit design.

Loop system. - A loop circulation system was analyzed for the blade configuration shown in figure 4 with two cooling mediums, water and fuel. A range of water flows entering the turbine (0.03 to 0.10 lb/sec-blade) and a single fuel flow (0.10 lb/sec-blade) were considered at sea-level

static conditions at an inlet-gas temperature of 2040° F. The trends that could be expected over a range of coolant flows with the loop circuit are shown for water and these trends would be similar for most liquids, including fuel. The single flow rate chosen for the fuel calculations was in the approximate range required for afterburning and therefore represents the most favorable flow condition for a fuel-cooling system. Values of blade-to-coolant heat-transfer coefficients, over-all coolant-temperature rise, average coolant temperature, and
2452 mean outside-surface blade temperatures obtained are compared in figure 7. At a coolant flow of 0.10 pound per second per blade, which corresponds to a coolant-flow ratio of 0.0367 at sea-level static conditions, figure 7 indicates a slightly higher (less than 60° F) mean outside-surface blade temperature, average coolant temperature, and over-all coolant temperature rise with fuel than with water. Comparison of the heat-transfer coefficients obtained for fuel with the straight-through and the loop systems at a flow rate of 0.10 pound per second per blade indicates that the loop circuit provides a coefficient approximately ten times larger than the straight-through circuit. The loop circuit does not afford an advantage to the fluids considered, even though the heat transfer is improved, because adequate heat-transfer rates are obtainable with the straight-through system. The loop circuit probably is best applicable to cooling systems utilizing high-boiling-point fluids which have heat-transfer characteristics inferior to those of water because the loop circulation provides a method of increasing heat-transfer rates. The loop circuit may also be applied for the case where a relatively small difference between the prevalent blade temperature and the coolant-discharge temperature is required.

Water-Cooling-System Characteristics

In the comparison made in this analysis between the water- and fuel-cooling systems, both fluids were considered to be in the same turbine-blade configuration and at the same coolant flow; the influence of the physical properties was thus emphasized. Unless specifically noted, all the results presented were calculated for a straight-through cooling system. The water-cooling system will be discussed first followed by a comparison with the fuel system.

Heat-rejection rates. - The heat-transfer coefficient from gas to blade is primarily influenced by the engine mass flow and the ratio of gas-to-blade temperature. Table II indicates that the compressor weight flow at sea-level static conditions was 158 pounds per second and that at an altitude of 35,000 feet and a Mach number of 1.8, the compressor weight flow reached a maximum of 189.7 pounds per second and decreased to 72.4 pounds per second at the combat Mach number of 1.8 and altitude of 50,000 feet. This large variation results in similar variations in the turbine-blade Reynolds number and gives the average profile heat-transfer coefficient which is shown in table III. Although the blade

design and engine operating conditions for the water- and fuel-cooled turbines are the same, the average outside heat-transfer coefficients differ because of the difference in blade-wall temperature, which has a strong influence on the coefficient. Except for a brief instant during take-off, the gas-to-blade heat-transfer coefficient for the water-cooled turbine varies from a minimum of 111 Btu/(hr)(sq ft)($^{\circ}$F) to a maximum of 244 Btu/(hr)(sq ft)($^{\circ}$F) during the acceleration from Mach number 0.8 to 1.8 at an altitude of 35,000 feet. The conditions imposed on the cooling system, therefore, change very rapidly at some points in the mission since the time required for this acceleration is less than 1.5 minutes. 2452

The rate of heat rejection to the coolant is an important design criterion, and for the water-cooled turbine at an altitude of 35,000 feet, the heat-rejection rate is a minimum of 359 Btu per second and a maximum of 704 Btu per second at Mach numbers of 0.8 and 1.8, respectively. Although the magnitude of the heat-rejection rate at the altitudes investigated is greatest at supersonic speed, it can be seen from the ratio of heat transferred to turbine work and the coolant temperature rise (table III) that the most critical point in the mission with respect to the rotor-blade heat rejection is the subsonic Mach number of 0.8 at an altitude of 35,000 feet. At this flight condition, the heat transfer to the coolant is 6 percent of the turbine work and the coolant-temperature rise increases to 201° F. This results from the unequal rates of change of outside and inside heat-transfer coefficients as flight conditions vary even though the ratio of coolant-to-gas flow is held constant. In spite of the large variations that occur in the operating conditions and heat-rejection rates, the average outside-surface blade temperature shown in table III remains stable and varies only from 636° F to 761° F. These temperatures are adequate on the basis of maximum stress limitation as determined from yield-strength characteristics of the material and maximum centrifugal blade stress which occurs at the blade base (fig. 5). The low blade temperatures obtained also indicate that materials such as low-alloy-content steels are probably applicable to a water-cooled turbine, thus reducing strategic material requirements. Since the water-cooling system is a closed circuit with a considerable volume of fluid, the rapid changes in operating conditions up to the combat Mach number and altitude can probably be absorbed readily. Consequently, the operating condition at a Mach number of 1.8 and an altitude of 50,000 feet, which represents by far the longest phase of the flight plan, was selected as the design point for the water-coolant radiator.

Radiator design. - The design conditions and approximate characteristics of a water-coolant radiator for the liquid-cooled turbojet engine under consideration are given in table IV. Following preliminary calculations of heat-rejection requirements and velocities in the radiator tubes at combat Mach number and altitude, the radiator frontal area was

established as 3.5 square feet for a core section 12 inches deep, composed of 0.230-inch-outside-diameter tubes extruded to a hexagonal shape at the tube ends and with a dimension of 0.260 inch across the flats. The characteristics of this core section, which were assumed to apply for the conditions of this analysis, are given in reference 18 and in figure 8. The ram-air temperature at the core face to which heat is ultimately rejected was 186° F. Consequently, the radiator water-out temperature (turbine-coolant-inlet temperature) which was originally assumed to be 150° F at all points in the mission was not adequate for this application. The total heat-rejection rate and coolant temperature rise given in table III for a turbine-coolant inlet temperature of 150° F were then recalculated for a value of 200° F. For the latter condition, only slight decreases in heat-rejection rate and temperature rise occurred, $1\frac{1}{2}$ and $2\frac{1}{2}$ percent, respectively. These values were then used to determine the heat-rejection rate of the radiator core per unit frontal area per 100° F inlet-temperature difference and are given in table IV. This table shows that the maximum turbine-coolant discharge temperature was found to be 395° F, corresponding to a saturation pressure of 233 pounds per square inch absolute, to prevent boiling at any point in the mission. The heat-rejection rate at design conditions was 403 Btu per second, or at the rate of about 572 horsepower. The ram-air flow, determined from figure 8 for this heat-rejection rate, was 16.10 pounds per second or about 22.2 percent of the engine mass flow. From this it can be seen that efficient thrust recovery in the ram-air duct is essential to avoid a large drag loss. The operating points of the radiator for a 200° F turbine-coolant inlet temperature at other points in the mission are also given in figure 8 and table IV. The required ram-air mass flow through the radiator is excessive at a Mach number of 1.8 and an altitude of 35,000 feet because of the low temperature difference between the radiator inlet and ram air coupled with the high heat-rejection rate. As previously mentioned, however, this operating condition is encountered for a few seconds during acceleration and would probably result in a momentary increase in water temperature of a magnitude determined by the heat capacity of the entire volume of water in the cooling system. The most effective way of reducing the radiator size and decreasing the ram-air flow is to increase the turbine-coolant inlet temperature, which will result in lower heat-rejection rates and in increased temperature difference between the radiator water and the ram air. With water, this could result in extreme pressures; therefore, the indicated solution is to use a high-boiling-point liquid such as Dowtherm-A. The resultant increase in blade temperature would probably also be desirable on the basis of heat rejection and thermal stresses in the blade.

An estimate of the radiator weight and drag was made to evaluate the over-all suitability of the water-cooling system. The weight of the filled radiator body, based on use of aluminum tubes, for one engine

was found to be 184 pounds, which amounts to about 6.6 percent of the basic engine weight. The drag of the radiator duct was calculated on the basis of 70 percent ram pressure recovery at the core face and consideration of pressure losses according to reference 20. It was found to be about 60 pounds at combat conditions. The net drag of the radiator duct is about 0.88 percent of the engine thrust for this particular flight condition. The drag, however, could become very large during acceleration at an altitude of 35,000 feet if the ram-air flow were increased to maintain a 200° F turbine-coolant inlet temperature. This is indicated by the heat-rejection rate shown in table IV in the column corresponding to a Mach number of 1.8 for an altitude of 35,000 feet.

General applicability. - An analysis of the water-cooling system indicates that the highly effective blade cooling coupled with the relatively ineffective final heat rejection presents the principal problem. The severe transient conditions encountered at some points in the mission can probably be met by the heat capacity of the volume of water in the system and these conditions do not constitute a significant limitation.

The ability of the water-cooling system to accommodate increasingly severe conditions is limited by heat-rejection considerations and excessive pressures. The engine installation in the supersonic interceptor aircraft used as a basis for this analysis was designed to permit a continuous 2g combat maneuver at a Mach number of 1.8 and an altitude of 50,000 feet, the design point for the cooling systems considered. The thrust available at the design point would also permit this aircraft to achieve a Mach number of approximately 2.5 in level flight. At this condition the ram temperature at the radiator core face would reach about 423° F. The coolant pressures required to permit heat rejection to ram air then become impractical and make the water-cooling system with heat rejection to ram air inadequate.

The design and installation problems of the water-cooling system are severe, as indicated by the necessity to dispose a 3.5-square-foot radiator weighing approximately 184 pounds or 6.6 percent of the basic engine weight in a suitable duct for efficient thrust recovery. This radiator frontal area is greater than half of the engine frontal area, an indication that a clean installation in a nacelle would be very difficult. The radiator duct must handle a ram-air flow of about 22.2 percent of the main-engine air flow at the combat Mach number and altitude, and although a design having less than 1-percent drag appears possible at steady combat conditions, severe drag losses would probably occur at off-design conditions, such as the critical conditions during acceleration to supersonic speed. For illustrative purposes a simple tube-type heat exchanger was utilized in this analysis. Application of a modern heat exchanger with extended heat-transfer surfaces would probably result in considerable reduction in frontal area, weight, and ram-air requirements. The mechanical complication of the water-cooling system

is evident since most of the plumbing, the pumps, and the radiator are located outside the main engine and are actually a part of the aircraft. These parts must be disconnected when the engine is removed for maintenance. Vulnerability of such a system is also evident.

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Interpretation of the results of this analysis indicates that the water-cooling system could only be applied in large fuselage engine installations at relatively low flight Mach numbers where the problem of installing a radiator of large frontal area is less severe and where the radiator duct can be located close to the engine so as to minimize weight and plumbing. The fact that a water-cooling system can be applied to nonafterburning engines suggests its possible application in transonic turbojet bombers. Application in supersonic interceptor aircraft of the type considered in this analysis does not appear promising in view of the operating limitations and degree of complication.

Fuel-Cooling-System Characteristics

Heat-rejection rates. - A comparison of fuel and water coolants is given in table III. As a result of the higher blade temperatures encountered with fuel cooling, the average outside heat-transfer coefficients and heat-rejection rates for the fuel-cooling system are lower than those for the water-cooling system. Except for the interval during take-off, the maximum heat-rejection rate encountered with the fuel-cooling system is 533 Btu per second during acceleration to a Mach number of 1.8 at an altitude of 35,000 feet, which is 24 percent less than with water cooling. The maximum fuel-temperature rise occurs at a Mach number of 0.8 and an altitude of 35,000 feet, resulting in a turbine-coolant discharge temperature of approximately 401° F, based on a fuel-tank temperature of 100° F and an allowance of 50° F for temperature rise in the fuel pumps. The minimum permissible discharge pressure for this condition so that fuel boiling will be prevented is approximately 280 pounds per square inch absolute as compared with 233 pounds per square inch absolute for the water-cooling system. The mean outside-surface blade temperature for the fuel-cooling system varies from 944° to 970° F, and it is evident that the afterburner fuel flow is adequate for turbine cooling under the conditions considered in this analysis. Since the afterburner fuel-air ratio remains approximately constant over the mission, there is no particularly critical flight condition for the fuel-cooling system, and stable operation can be expected provided adequate pressure is maintained in the system to prevent boiling at high fuel temperatures.

Tank recirculation for nonafterburning operation. - The necessity for the provision of loiter or cruising time at altitudes has been previously discussed in the description of the fuel-cooling system. This is accomplished by rejecting heat to the large reserve of fuel in the

tanks during the early portions of the mission and maintaining the fuel flow through the turbine-cooling system at approximately the value ordinarily used in the afterburner. The variation of total fuel weight and fuel-tank temperature with time for the case of heat rejection to the fuel tanks at the cruising flight conditions (Mach number, 0.8; altitude, 35,000 ft) is shown in figure 9. The initial fuel weight was 8500 pounds and the initial fuel temperature was assumed to be 100° F. The heat-transfer rates used to obtain the solid lines for a turbine-inlet temperature of 2040° F were taken from table III. The rate of 2452
temperature increase of the fuel is dependent upon the rate of heat addition to the fuel and the weight of fuel in the tanks. The circled points in figures 9(a) and 9(b) represent the tank depletion and the fuel temperature after the climb to cruise conditions. The climb was made with afterburner on so that no heat was rejected to the tanks for the first 110 seconds. Cruising can be carried out for 320 seconds (about 5 minutes) if the turbine-inlet temperature is 2040° F and the limiting tank temperature is 150° F. This tank temperature corresponds to a differential pressure between fuel tank and ambient pressure of about 13.5 pounds per square inch to prevent evaporation losses. As previously mentioned, the aircraft was overpowered for cruise at these flight conditions and it was then assumed that a reduction in turbine-inlet temperature from 2040° to 1620° F was necessary to permit subsonic cruising at an altitude of 35,000 feet. This temperature reduction resulted in a 50-percent decrease in heat-rejection rate to the tanks, assuming the blade temperature held constant. The tank-fuel depletion rate was reduced because of the lower fuel-air ratio. As shown by the dashed lines in figure 9, the permissible cruise time increases to 615 seconds (about 10 minutes) and the fuel remaining in the tanks at the end of the permissible time was 5400 pounds. The combat time would, of course, be considerably less following an extended cruise period.

The fuel-cooling system with heat rejection to the fuel tanks appears to provide sufficient flexibility with respect to provision for loiter time at an altitude of 35,000 feet although pressurization of the fuel tanks to avoid excessive evaporation losses is necessary. If desired, both climb and cruise portions of the flight can be made without afterburning, in which case the cruise time is decreased because of the rise in tank-fuel temperature during climb.

General applicability. - Fuel cooling should be considered only for an afterburning engine where sufficiently high fuel-flow rates are available. The principal limitation in the fuel-cooling system is the need for provision of periods of nonafterburning operation at cruising speed and altitude. If the fuel-cooled system is to be applicable during extended periods of nonafterburning operation, provision must be made for adequate pressurization of the fuel tanks or refrigeration of

the fuel supply prior to take-off in order to avoid excessive evaporation losses. The need for pressurization indicates that the system could only be applied when the aircraft configuration permits the location of fuel tanks in the fuselage so that simple pressure vessels can be designed; the pressurization of wing tanks is limited by the relatively large flat unsupported surfaces. The most promising application for the fuel-cooling system is in the turbojet guided missile at high flight Mach number in which continuous afterburner operation is required;
2452 thus, heat rejection to the fuel tanks is avoided.

The fuel-cooling system appears able to accommodate increasingly severe operating conditions because the temperature of the heat-rejection medium is less dependent upon ram temperature than that of the water-cooled system. This system is therefore relatively insensitive to high flight Mach number. An uncertain limitation in the application of fuel-cooling systems is the chemical stability of the fuel. Although the average temperature of the fuel at turbine discharge does not exceed 401° F, higher local temperatures may be encountered in the blade coolant passages where the passage wall is much hotter.

The design, installation, and mechanical problems of the fuel-cooling system are relatively simple and weight and performance penalties appear negligible. Since the fuel-cooling system requires only that the afterburner fuel be passed through the turbine prior to injection, there is no additional plumbing necessary outside the main engine except for a return line to the fuel tank for nonafterburning operation. This represents a considerable simplification over the water-cooling system. The fuel-cooling system requires no extra controls since the coolant-flow ratio is determined by the afterburner fuel-air ratio. Internal pressures required to prevent local boiling in the system appear comparable with present practice. The required pressure is readily provided by the thermosiphon pumping action in the turbine-cooling system and would otherwise be necessary for fuel injection in the afterburner nozzles. Some hazard might result from the high temperature of the fuel emerging from the turbine rotor, and the increased vulnerability of pressurized fuel tanks is a factor to be considered.

Regenerative Refrigeration Liquid-Cooling-System Characteristics

The regenerative refrigeration liquid-cooling system is a heat-pump cycle operating off the main-engine cycle in which the energy level of the coolant must be raised sufficiently high to reject the turbine-cooling load back into the main-engine working fluid at compressor discharge. There are two distinct variations of this system dependent on temperature and pressure of the turbine-coolant discharge. In the first case, the saturation temperature of the coolant vapor is below the compressor-discharge temperature and an external vapor compressor is required to operate the cycle. In the alternate case, coolant-vapor

saturation temperature is higher than compressor-discharge temperature; as a result, the thermosiphon pumping action in the rotor provides sufficient pressure to operate the cycle.

A complete survey of various fluids which may be applicable to a regenerative refrigeration system has not been made and it is not known what fluid would prove most advantageous. Although water would be an impractical fluid to use for a regenerative system because of the very high pressures that would be required, a simple analysis of a low-temperature water-vapor cycle was made to illustrate the mode of operation with external compression. The main-engine compressor-discharge temperature at combat Mach number and altitude, given in table II, is 604° F, which is also the minimum temperature at which a liquid coolant can be taken from a condenser located in the main working fluid at the discharge of the compressor. It should be noted that the temperature difference between compressor-discharge air and the condensing fluid should be fairly large to minimize the condensing surface required. For purposes of illustrating the regenerative system, this temperature difference was assumed to be nearly zero in order to determine the operating characteristics for the minimum permissible coolant pressures. The following assumptions, summarized in table V, were made: turbine-heat-rejection rate, 250 Btu per second; turbine-coolant-inlet temperature, 400° F; turbine-coolant-inlet pressure, 247 pounds per square inch absolute; mean outside-surface blade temperature, 1060° F. The heated water emerging from the turbine at high pressure was throttled to the saturation temperature of 400° F and pressure of 247 pounds per square inch absolute, resulting in 0.302 pound per second of vapor at 400° F. The vapor, which represents the heat rejected in the turbine, is then passed into the refrigeration cycle, compressed, and passed into the condenser. The temperature of the liquid leaving the condenser and entering the expansion valve is about 605° F, corresponding to a saturation pressure of 1600 pounds per square inch absolute.

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The quantity of liquid desired from the expansion at a temperature of 400° F is 0.302 pound per second, so that the liquid originally taken out of the turbine as a vapor can be replaced. For the conditions of this calculation, 70 percent of the fluid entering the expansion valve is recovered as a liquid; therefore, the total rate of circulation through the refrigeration cycle must be 0.432 pound per second and the residual flash vapor, which amounts to 0.130 pound per second or 30 percent of the total, must be returned to the refrigeration cycle.

Regeneration of the turbine-cooling load required a shaft power extraction of approximately 150 horsepower for the refrigeration compressor; this amounts to about 1.5 percent of the turbine output. The heat-rejection rate in the condenser amounts to 356 Btu per second and includes both the heat rejected in the turbine and the refrigeration compressor power. The energy expended by the refrigeration compressor

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is thus partly returned to the cycle by heat rejection to the working fluid in the condenser, but the net effect on specific fuel consumption was not estimated. The principal problems brought out by consideration of the water-vapor cycle are the following: the extreme pressures required to permit condensation to compressor discharge air, the large pressure ratio of the auxiliary compressor, and the relatively large residual flash vapor that must be recirculated in the cycle. Study of the vapor-pressure characteristics of possible alternate fluids, illustrated in figure 10, shows that a system with greater advantages than the water cycle could possibly be achieved. The first objective in selecting a new fluid other than water is to increase the temperature level of the coolant throughout the cycle and at the same time find a coolant that will satisfy both the heat-transfer and refrigeration characteristics required for a regenerative refrigeration cycle. The coolant must be stable so that there will be no chemical breakdown that would cause a change of fluid properties or that would leave deposits in the coolant system as the coolant is evaporated. Curves 3, 4, and 5 in figure 10 are for liquids with higher boiling points than the coolant investigated. If other properties were acceptable, these liquids would permit operation in the desired temperature range with moderate pressures. If Dowtherm-A were considered, the vapor condenser pressure would be reduced to less than 100 pounds per square inch absolute and the external refrigeration compressor could possibly be eliminated because the thermosiphon pumping action of the turbine can supply sufficient pressure. Table II indicates, however, that at an extreme flight Mach number such as the level flight speed previously cited, the main compressor-discharge temperature is 802° F. Thus, in a regenerative system, condensation must occur above 802° F, and even with high-boiling-temperature liquids such as Dowtherm-A, an auxiliary compressor may still be required.

An additional advantage of higher coolant temperature is the reduced heat-rejection rate from the turbine with consequent reduction of size and weight throughout the system. The analysis given previously is only suggestive of the possibilities of regenerative liquid systems, but it appears that if a suitable fluid were found for this type of system, it would be possible to operate such a system with moderate refrigeration horsepower at the combat Mach number of 1.8 and altitude of 50,000 feet.

General applicability. - The fuel-cooling system previously discussed was found to be suitable only for afterburning engines. The water-cooling system was shown to have severe limitations. The only remaining possibility for a liquid-cooling system with application to high-thrust nonafterburning turbojets at supersonic flight speed is regeneration of the turbine-cooling load into the main-engine working fluid at the compressor discharge. Although the regenerative system appears mechanically complicated, it has the singular advantage of being entirely contained within the engine and capable of operation without

external systems in the aircraft. This represents a considerable advantage in engine installation and maintenance and provides a "package" installation that is entirely controlled by the engine manufacturer. Since the coolant flow through the turbine rotor can be varied independently, the system is flexible in operation. Provision of higher coolant-discharge temperatures through the use of high-boiling-point liquids is a promising method of eliminating the need for an auxiliary compressor and reducing the heat-rejection rates required. The principle problem, in further consideration of the regenerative system, appears to be selection of a suitable fluid that has both the desirable heat-transfer properties and vapor-pressure characteristics in the desired range.

RESULTS AND CONCLUSIONS

The following results, which indicate the general limitations encountered in liquid-cooled turbojet engines at supersonic speed and high altitude, were obtained from an anlytical investigation of several turbine-blade liquid-cooling systems designed for a turbojet-powered interceptor aircraft:

1. For a coolant-flow ratio of 0.03 the mean outside-surface blade temperatures obtained by use of water and fuel coolants were approximately 653° and 944° F, respectively. These temperatures are considered adequate with respect to maximum blade centrifugal-stress limitations for the type and size of engine currently of interest for interceptor application.

2. For the design point at combat conditions, the water-cooling system required a heat-rejection rate equivalent to approximately 572 horsepower; a radiator weight of 184 pounds, or about 6.6 percent of the basic engine weight; a radiator frontal area of 3.5 square feet, or greater than half of the basic engine frontal area; a ram-air flow of 22.2 percent of the basic engine-air flow; and a drag of less than 1 percent of the basic engine thrust. The analysis indicates that a water-cooling system can be applied to large fuselage engine installations at relatively low flight Mach number, but the system is inadequate for the supersonic interceptor aircraft mission used as the basis of this analysis.

3. The fuel-cooling system required a maximum heat-rejection rate which was 24 percent less than that for the water-cooling system and can be applied in the interceptor if provision for pressurization of the fuel tanks is made. Up to 10 minutes of nonafterburning cruising operation with heat rejection to pressurized fuel tanks during initial portions of the flight could be provided for the particular mission and engine design considered in the analysis.

4. The fuel-cooling system is considered promising because weight and performance penalties appear negligible; design, installation, and mechanical problems appear relatively simple; and the system is insensitive to high flight Mach number. The most promising application is to a turbojet guided missile which employs continuous afterburning.

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5. A regenerative liquid-cooling system appears capable of operation over the desired range of flight conditions without an external radiator, provided that suitable fluids are found. Such a system can be applied to nonafterburning as well as afterburning engines and provides the advantage of a "package" installation entirely contained within the engine and capable of operation without external systems in the aircraft.

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National Advisory Committee for Aeronautics
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APPENDIX A

SYMBOLS

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The following symbols were used in this report:

A	cross-sectional area through blade coolant passage, sq ft
b	blade height or span, ft
c	specific heat of liquid, Btu/(lb)(°F)
c_p	specific heat of gas at constant pressure, Btu/(lb)(°F)
D	hydraulic diameter of blade coolant passage, ft
$\bar{F}$	theoretical coefficient of Reynolds number in heat-transfer relation
f	fuel-air ratio
g	acceleration due to gravity, ft/sec^2 (32.17)
H	average heat-transfer coefficient, Btu/(sec)(sq ft)(°F) unless otherwise noted
h	enthalpy, Btu/lb
$h_{l,v}$	heat of vaporization, Btu/lb
k	thermal conductivity, Btu/(sec)(sq ft)(°F/ft)
l	blade perimeter, ft
N	number of pairs of cooling passages
Nu	Nusselt number
Pr	Prandtl number
Q	heat-transfer rate, Btu/sec
q	heat transfer per unit weight of combustion-gas flow, Btu/lb
Re	Reynolds number
S	surface area, sq ft

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T temperature, $^{\circ}F$

W relative velocity, ft/sec

w weight flow of fluid, lb/sec

w' weight, lb

X fraction of liquid after throttling in liquid-cooled regenerative system

Δy thickness of blade material for heat transfer, ft

Z theoretical exponent of Reynolds number in heat-transfer relation

μ absolute viscosity of fluid, (lb)(sec)/sq ft

ρ density of fluid, slugs/cu ft

τ time, sec

Ω specific turbine work, Btu/lb, (work per unit weight of combustion-gas flow)

Subscripts:

av average

B blade

C engine compressor

e effective

f fuel

g combustion gas

i inside blade surface

l cooling liquid

m mean

o outside blade surface

v cooling liquid evaporated

0 NACA sea-level air or initial conditions of tank fuel

1 liquid-coolant inlet to turbine

2 liquid-coolant outlet from turbine 2452

3 before expansion valve

4 after expansion valve

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APPENDIX B

DERIVATION OF EQUATION FOR CALCULATING TEMPERATURE RISE OF TANK FUEL WHEN USED AS HEAT-REJECTION MEDIUM

The rate of heat transfer to the blades is assumed to be known from the characteristics of the cooling system and can be expressed in terms of the tank-fuel weight as follows:

$$Q = w'_f c \frac{dT_f}{d\tau}$$

$$Q = c\left(w'_{f,0} - \int_0^\tau f w_C \, d\tau\right) \frac{dT_f}{d\tau}$$

where

w'_f tank-fuel weight, lb

$w'_{f,0}$ initial tank-fuel weight, lb

w_C engine-air flow rate, lb/sec

c specific heat of fuel

$$\frac{dT_f}{d\tau} = \frac{Q}{c\left(w'_{f,0} - f w_C \tau\right)}$$

Integrating between the limits of time = 0 and time = τ and between the limits of initial and final tank-fuel temperature gives

$$T_f - T_{f,0} = -\left[\frac{Q}{c(f)w_C}\right] \ln\left(\frac{w'_{f,0} - f w_C \tau}{w'_{f,0}}\right) \tag{7}$$

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TABLE I - SUMMARY OF CONDITIONS AND ASSUMPTIONS

Flight conditions	
Mach number range	0 - 2.5
Altitude range, ft	0 - 50,000
Combat conditions	
Altitude, ft	50,000
Mach number	1.8
Maneuverability	2g
Loiter conditions	
Altitude, ft	35,000
Mach number	0.8
Basic engine specifications	
Sea-level specific mass flow, lb/(sec)(sq ft)	23.6
Turbine-inlet temperature, °F	2040
Sea-level compressor pressure ratio	6
Afterburning temperature, °F	3040
Aircraft specifications	
Gross weight, lb	28,000
Structure-to-gross-weight ratio	0.3
Pay load, lb	3000
Unaugmented specific engine weight, lb/lb thrust	0.28
Turbine specifications	
Tip speed, ft/sec	1500
Tip diameter, in.	35.1
Hub-tip ratio	0.732
Blade material	S-816
Blade critical section	1/3 span
Liquid-coolant passage diameter, in.	0.125
Pairs of coolant passages per blade	3
Blade chord, in.	2.27
Blade solidity	1.366
Blade span, in.	4.72
Number of blades	58
Assumptions for water-cooled system	
Coolant-flow ratio for straight-through circuit analysis	0.03
Coolant-flow range for heat-transfer coefficient comparison, straight-through circuit, lb/sec-blade	0.03 - 0.10
Coolant-flow range for loop circuit analysis, lb/sec-blade	0.03 - 0.10
Radiator ram pressure recovery	0.70
Radiator frontal area, sq ft	3.5

TABLE I - SUMMARY OF CONDITIONS AND ASSUMPTIONS - Concluded

Assumptions for fuel-cooled system	
Coolant-flow ratio for straight-through circuit analysis	0.03
Coolant-flow range for heat-transfer coefficient comparison with straight-through circuit, lb/sec-blade	0.03 - 0.10
Coolant flow for loop circuit analysis, lb/sec-blade	0.10
Initial tank-fuel temperature, $^{\circ}F$	100
2452 Temperature rise in afterburner fuel pump, $^{\circ}F$	50
Fuel temperature into turbine for afterburning case, $^{\circ}F$	150
Limiting tank-fuel temperature for case of heat rejection to fuel tanks, $^{\circ}F$	150
Assumption in regenerative liquid-cooling system	
Auxiliary compressor efficiency	0.80

TABLE II - ENGINE OPERATING CONDITIONS

Altitude, ft	0		35,000		50,000	
Flight Mach number	0	0.8	0.8	1.8	1.8	2.5
Compressor pressure ratio	6.0	5.2	7.0	4.6	4.6	2.6
Compressor efficiency	0.805	0.821	0.780	0.831	0.831	0.704
Compressor discharge temperature, °F	484	547	405	604	604	802
Compressor weight flow, lb/sec	158.0	199.3	58.6	189.7	72.4	82.7
Turbine-inlet temperature, °F	2040	2040	2040	2040	2040	2040
Afterburner temperature, °F	3040	3040	3040	3040	3040	3040
Ram temperature, °F	59	126	-17	186	186	423
Primary-burner fuel flow, lb/sec	4.06	4.92	1.58	4.53	1.73	1.73
Afterburner fuel flow, lb/sec	4.88	6.04	1.80	5.50	2.21	2.46
Turbine efficiency	0.83	0.83	0.83	0.83	0.83	0.83
Turbine pressure ratio	2.14	2.17	2.14	2.14	2.14	2.05
Specific fuel consumption, lb/lb-hr	2.11	2.48	2.14	2.09	2.14	2.40
Thrust, lb	15,283	15,950	5665	17,706	6793	6289

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TABLE III - COMPARISON OF FUEL AND WATER AS COOLANTS

[Turbine-inlet gas temperature 2040° F; turbine-coolant inlet temperature 150° F.]

Altitude, ft	0				35,000				50,000			
Flight Mach number	0		0.8		0.8		1.8		1.8		2.5	
Coolant	Fuel	Water	Fuel	Water	Fuel	Water	Fuel	Water	Fuel	Water	Fuel	Water
Coolant-flow ratio	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03
Coolant weight flow, lb/sec	4.74	4.74	5.98	5.98	1.76	1.76	5.69	5.69	2.17	2.17	2.48	2.48
Average inside heat-transfer coefficient, Btu/(hr)(sq ft)(°F)	751	2850	889	3320	329	1410	868	3240	414	1620	461	1800
Average outside heat-transfer coefficient, Btu/(hr)(sq ft)(°F)	206	216	241	252	107	111	233	244	124	128	135	141
Heat-rejection rate to coolant, Btu/sec	477	639	547	721	245	359	533	704	291	409	317	444
Ratio of heat transfer to turbine work, q/Ω	.0293	.0392	.0267	.0352	.0410	.0600	.0275	.0363	.0394	.0554	.0410	.0574
Coolant temperature rise, °F	178	134	163	120	251	201	166	123	232	188	221	177
Mean outside-surface blade temperatures	958	729	970	761	961	636	964	753	944	653	941	662

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TABLE IV - WATER-COOLANT-RADIATOR DESIGN CONDITIONS FOR HEAT REJECTION TO RAM AIR

[Core frontal area, 3.5 square feet; core length, 12 inches; turbine-coolant-inlet temperature, 200° F.]

Altitude, ft	0		35,000		50,000	
Flight Mach number	0	0.8	0.8	1.8	1.8	2.5
Turbine-coolant discharge temperature, °F	330	317	395	319	381	371
Turbine-coolant discharge saturation pressure, lb/sq in. abs	103	86	233	88	198	176
Ram-air temperature, °F	59	126	-17	186	186	423
Inlet-temperature difference, °F	271	191	412	133	195	-52
Heat-rejection rate, Btu/sec	628	709	353	692	403	437
Heat-rejection rate per unit frontal core area per 100° F inlet-temperature difference, Btu/(sec)(sq ft)(100° F)	66.2	106	24.5	148.7	59.1	---
Ram-air flow rate per unit frontal core area, lb/(sec)(sq ft)	5.2	9.2	2.0	15.2	4.6	---
Ram-air flow rate, lb/sec	18.2	32.2	7.0	53.2	16.1	---

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TABLE V - SUMMARY OF REGENERATIVE WATER-VAPOR REFRIGERATION SYSTEM

CHARACTERISTICS

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Mach number	1.8
Altitude, ft	50,000
Turbine-coolant inlet temperature, $^{\circ}F$	400
Turbine-coolant inlet saturation pressure, lb/sq in. abs	247
Mean blade temperature on outside blade surface, $^{\circ}F$	1060
Heat-rejection rate, Btu/sec	250
Refrigerant vapor from turbine-coolant discharge, lb/sec	0.302
Condenser temperature, $^{\circ}F$	605
Condenser pressure, lb/sq in. abs	1600
Residual flash vapor, lb/sec	0.130
Refrigerated liquid from expansion valve at condenser discharge, lb/sec	0.302
Total refrigerant cycle flow, lb/sec	0.432
Refrigerant compressor efficiency	0.80
Refrigerant compressor power, hp	150
Condenser heat-rejection rate, Btu/sec	356

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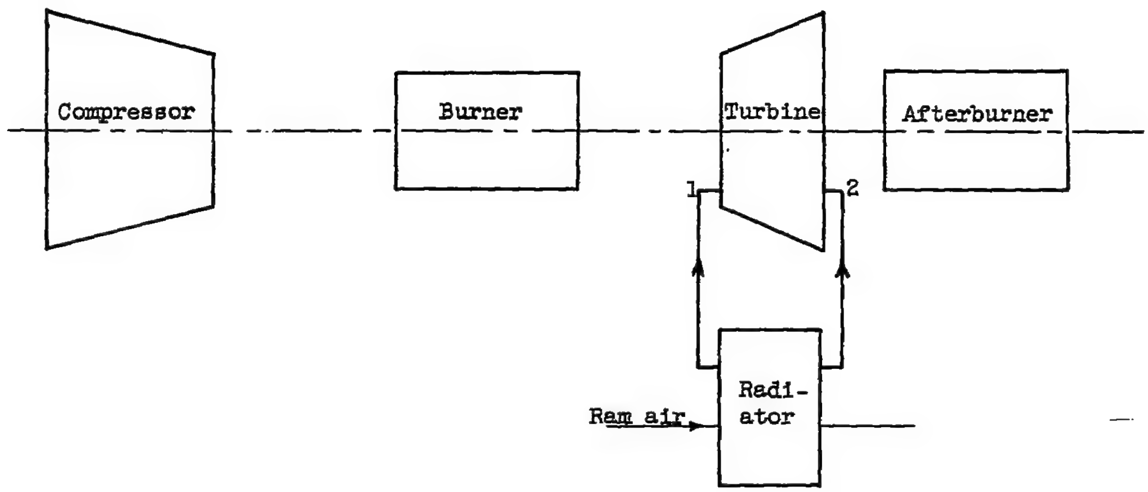

Figure 1. - Schematic diagram of water-cooling system with heat rejection to ram air.

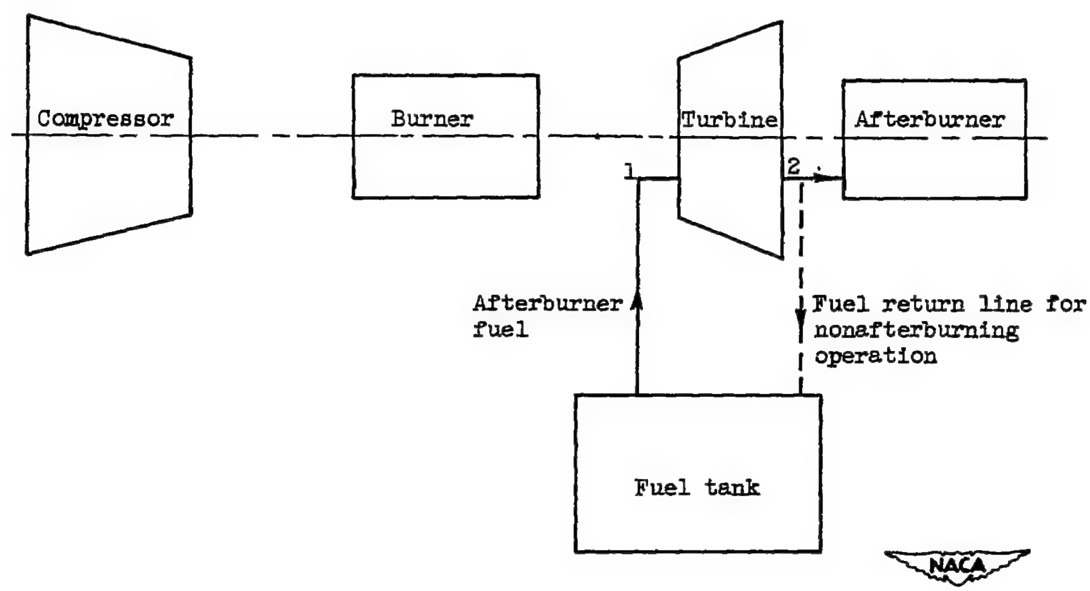

Figure 2. - Schematic diagram of fuel-cooling system with heat rejection to afterburner fuel. (Fuel recirculated to fuel tank for nonafterburning operation.)

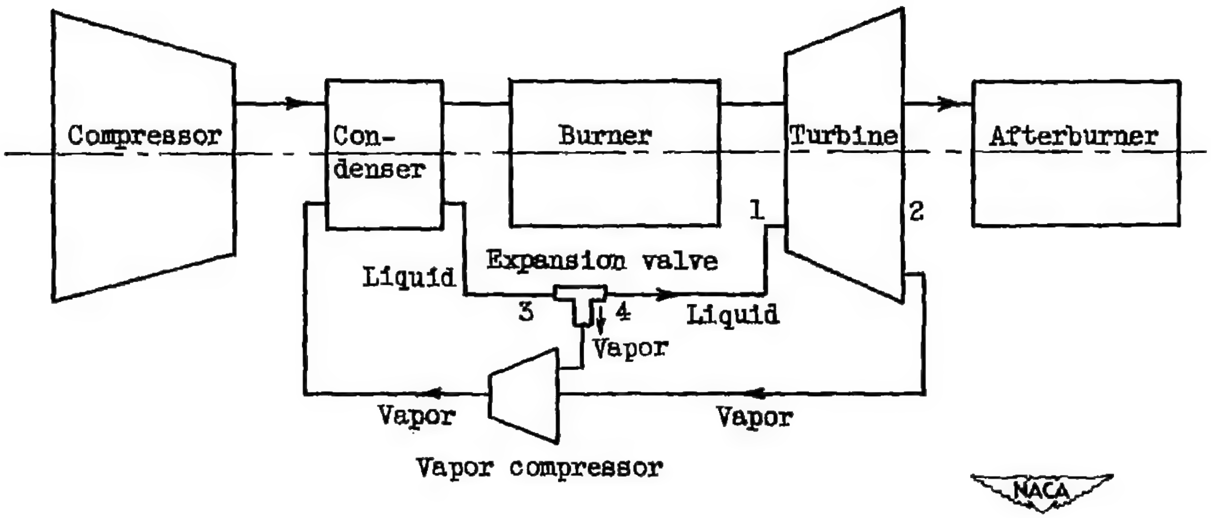

Figure 3. - Schematic diagram of regenerative liquid-cooling system with condensation of vapor in condenser located at compressor outlet.

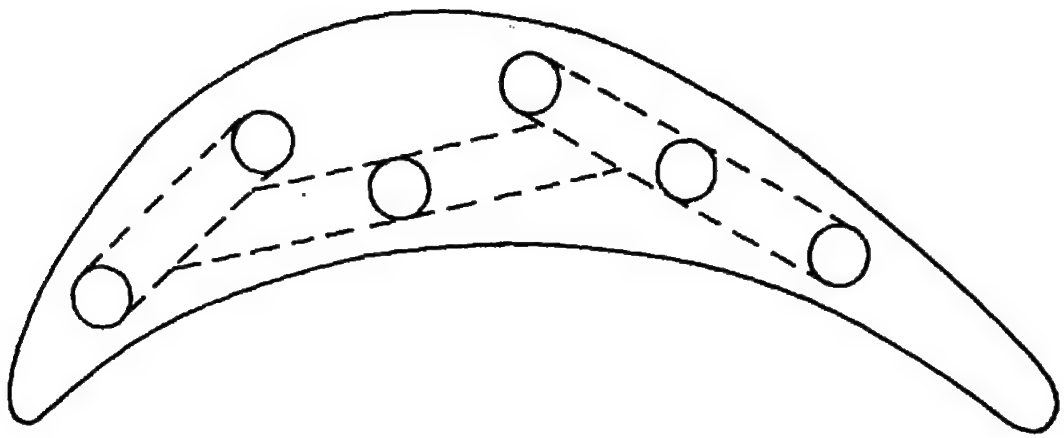

Figure 4. - Profile of constant-cross-section liquid-cooled blade used in analysis showing internal coolant-passage configuration. Dotted lines indicate cross-over passages at blade tip. Chord, 2.27 inches; coolant-passage diameter, 0.125 inch.

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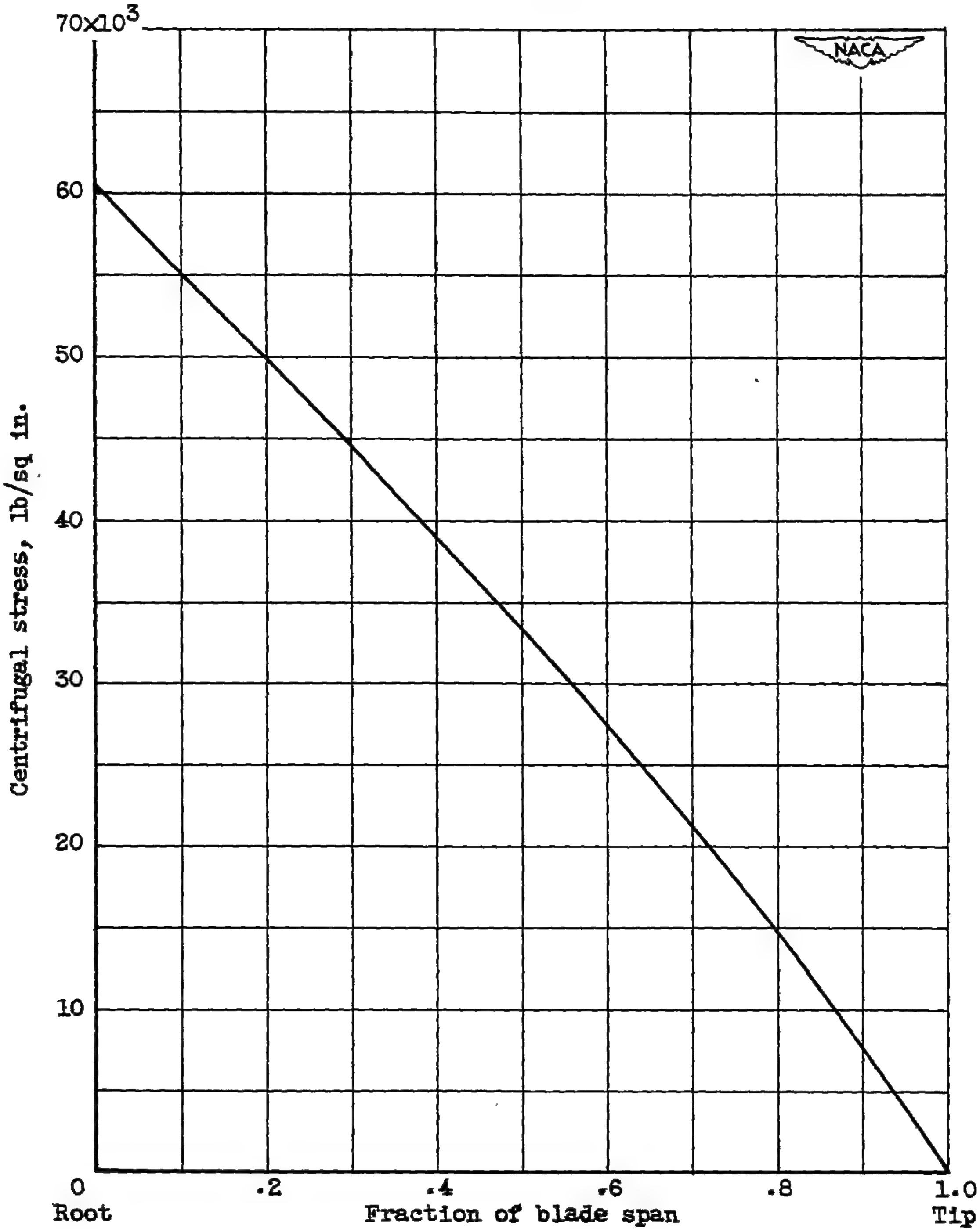

Figure 5. - Spanwise distribution of turbine-blade centrifugal stress for untapered blade. Blade material, S-816; hub-tip ratio, 0.732; blade span, 4.72 inches; tip speed, 1500 feet per second.

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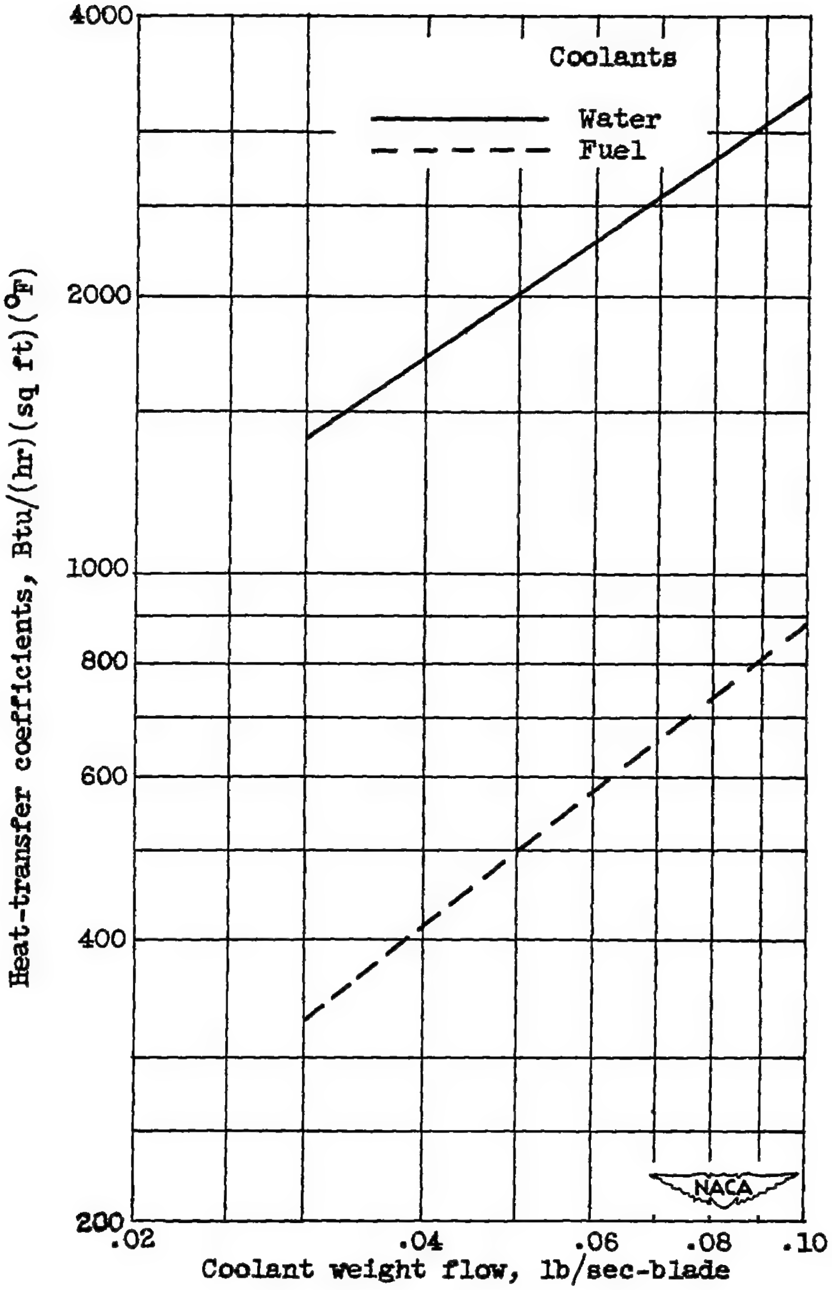


Figure 6. - Comparison of heat-transfer coefficients obtained with water and fuel in straight-through cooling systems. Coolant passages, 3 pairs of 0.125-inch-diameter holes.

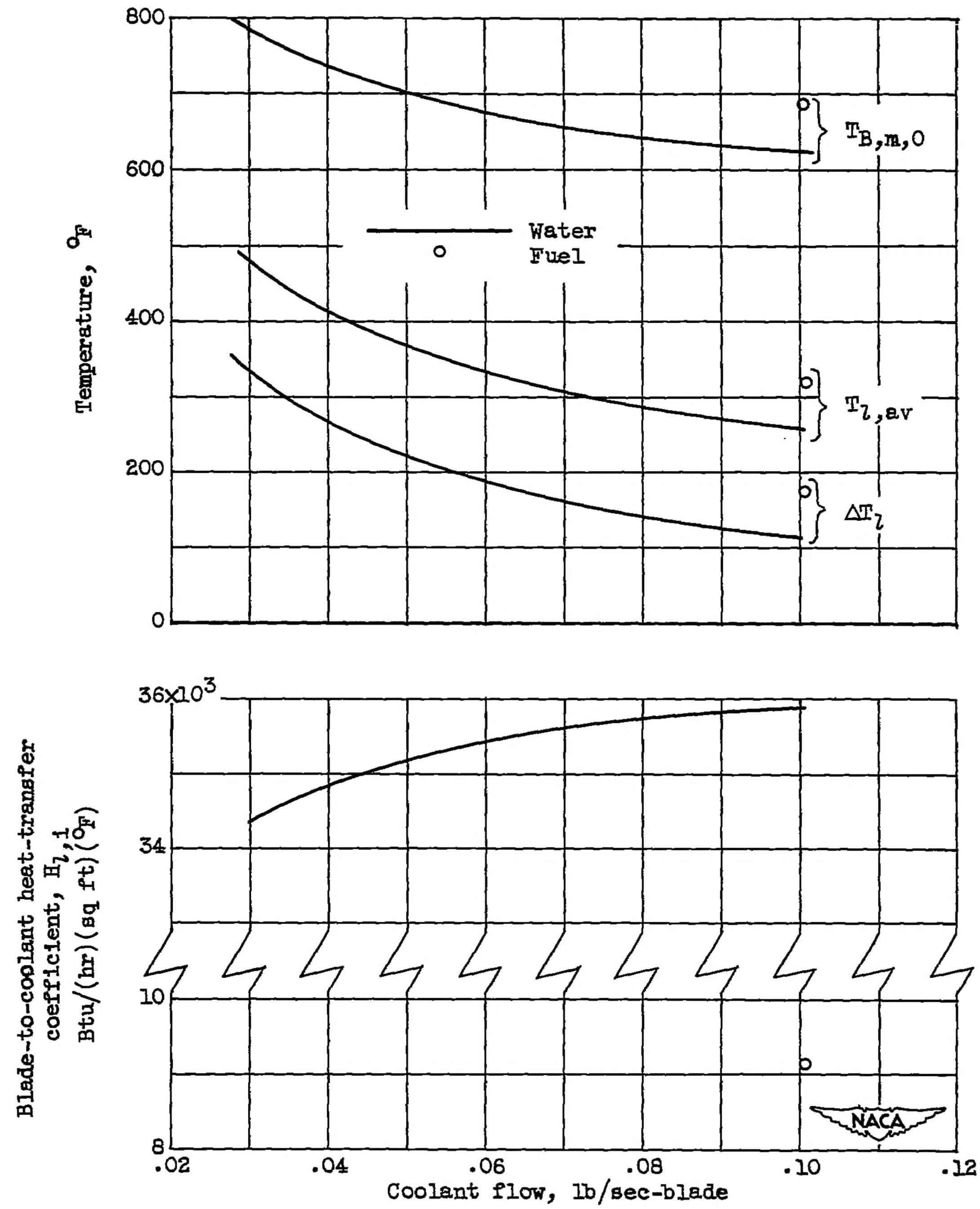


Figure 7. - Analysis of loop system over range of coolant flows for water and for one coolant flow for fuel showing mean outside-surface blade temperatures, average coolant temperatures, coolant temperature rise, and blade-to-coolant heat-transfer coefficients. Water properties evaluated at 300° F; fuel properties evaluated at 325° F; sea-level static conditions; effective gas temperature, 1760° F.

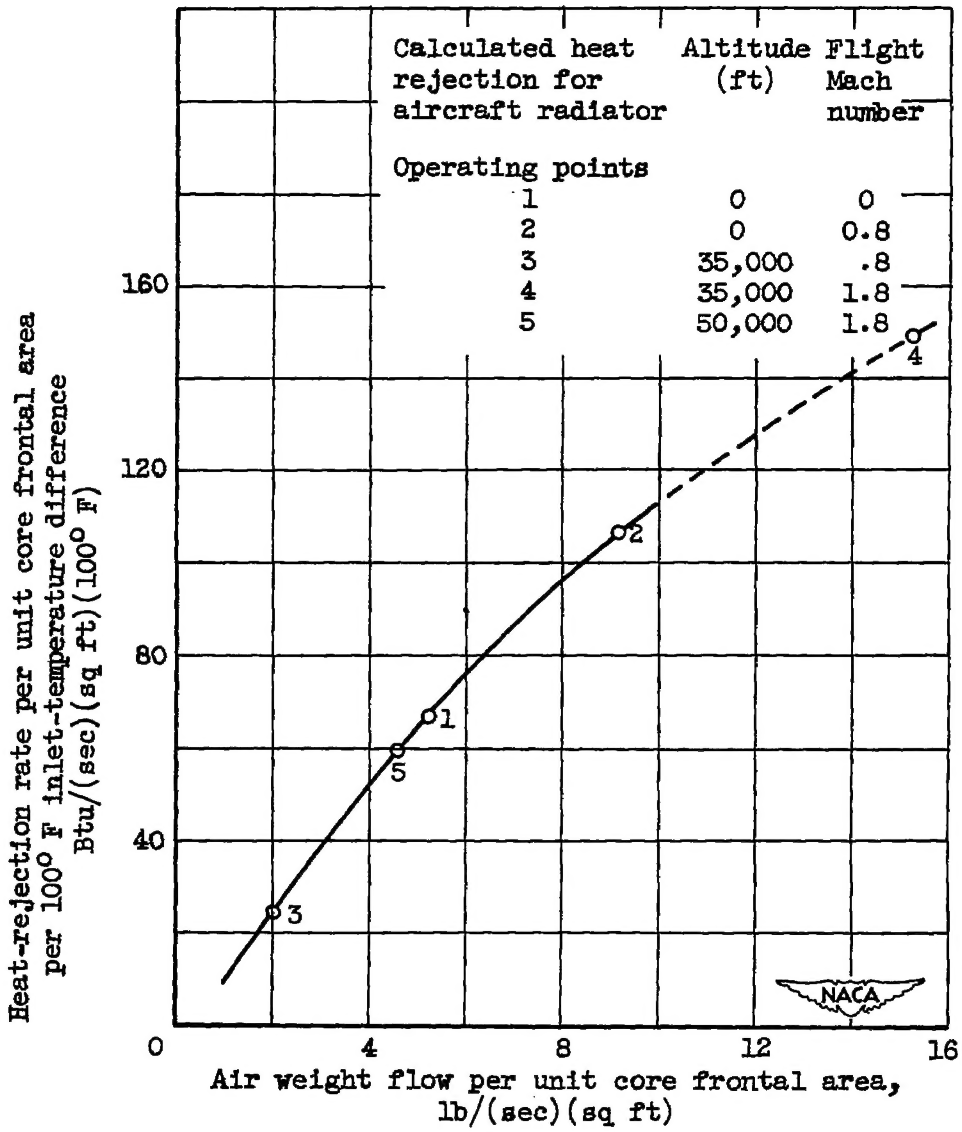
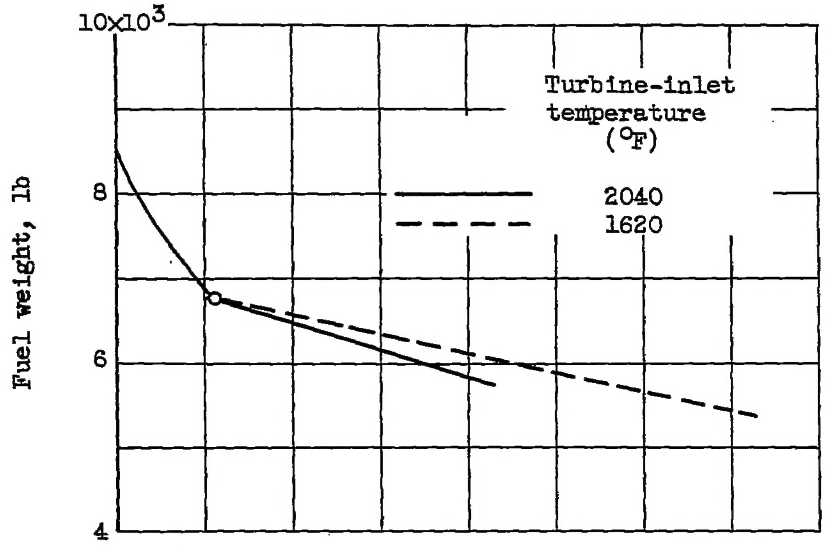
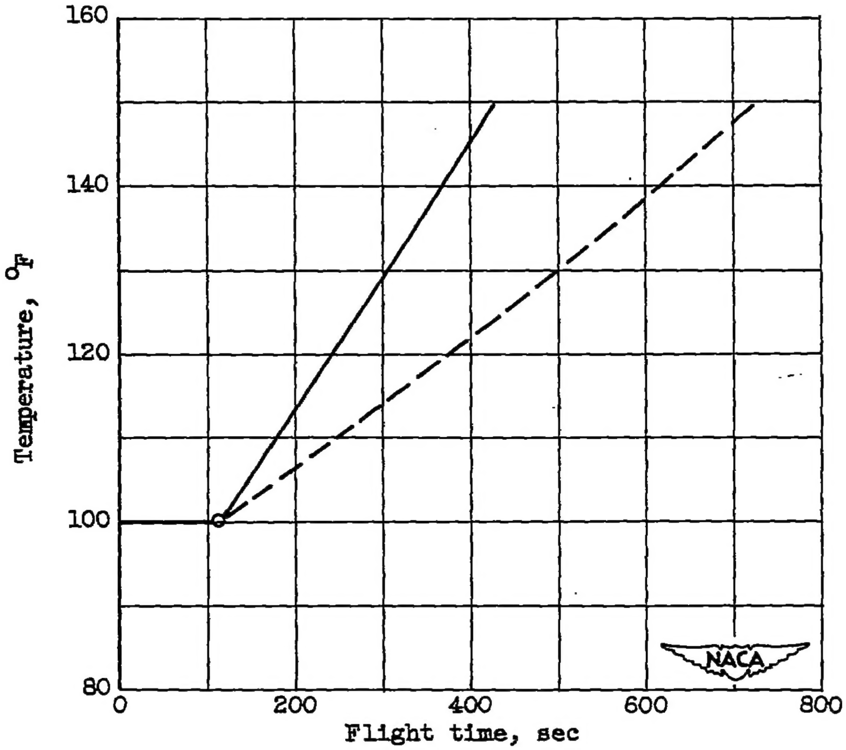


Figure 8. - Performance characteristics of a 12-inch core length of water radiator and calculated heat-rejection rates for aircraft radiator. Radiator frontal area, 3.5 square feet. Curve taken from reference 18.



(a) Quantity of fuel remaining in tank.



(b) Temperature of fuel in tank.

Figure 9. - Variation of fuel quantity and temperature in tank with flight time for two values of turbine-inlet temperature for heat rejection to fuel in tanks. Turbine-blade temperature, 961° F.

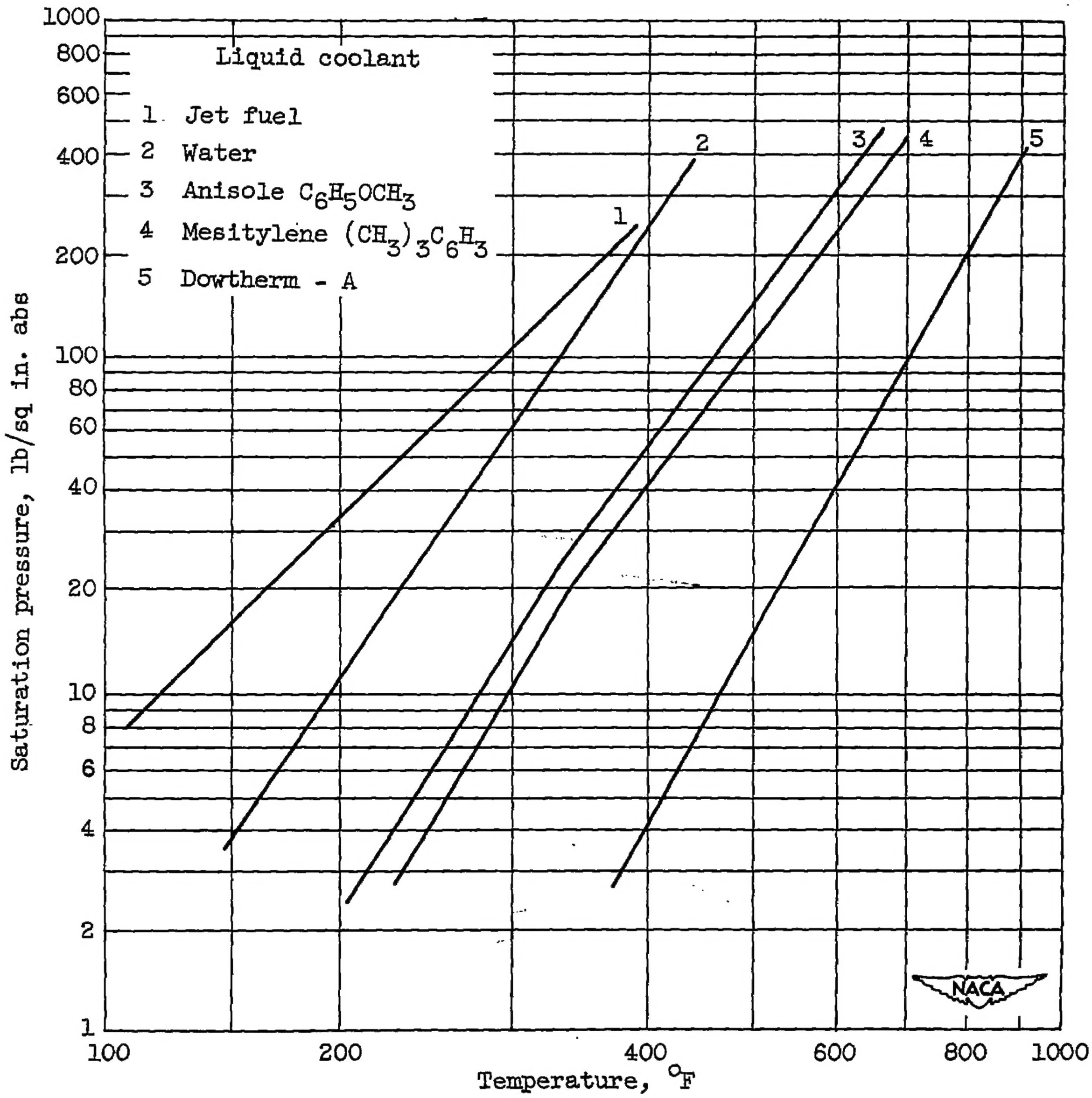


Figure 10. - Variation of saturation pressure with temperature of various liquid coolants. (Ordinate scale, logarithmic; abscissa scale, reciprocal.)

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